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THE DELTA AND THOR/AGENA LAUNCH VEHICLES FOR SCIENTIFIC AND APPLICATIONS SATELLITES

CHARLES R. GUNN

SEPTEMBER 1970

GSFC
GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

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ABSTRACT

The Delta and Thor/Agena medium class launch vehicles are described for potential users. Functional description of the vehicles, their performance, flight environment, organizational interfaces, spacecraft integration requirements, launch operations and costs are provided for Delta Model 904, and Thor/Agena Model 9A4. Projected vehicle growth currently under study is highlighted.

Both three stage vehicles utilize a common first and third stage: The first stage is the Universal Boattail (UBT) Long Tank Thor that can be thrust augmented with up to nine Castor II solid propellant motors; the third stage is the spin stabilized TE-364-4 solid propellant motor.

The Delta launch vehicle has been used by NASA, foreign governments, and U. S. private industry in over 80 launches of scientific and applications satellites. Since its inception Delta has undergone ten major upratings while maintaining an active, uninterrupted launch schedule and demonstrated high flight reliability and low cost. At least 40 more launches are scheduled over the next four years from the Eastern and Western Test Ranges. The Delta, Model 904, is composed of the UBT Thor booster with nine Castor II solids; the Delta second stage uprated with a new propulsion system and a strapdown inertial guidance system; and the TE-364-4 third stage. This new model of Delta is to be available in late 1971, costs about $5-1/2 million, and capable of injecting 4000 pounds into low earth orbit, or 1400 pounds into a geo-synchronous transfer orbit.

The Thor/Agena has been used by NASA and the U.S. Air Force in over 130 launches from the Western Test Range. Currently, however, there are no follow-on NASA Thor/Agena missions scheduled, though use by the U. S. Air Force is continuing. The Thor/Agena, Model 9A4, is composed of the UBT Thor booster with nine Castor II solids; the new Agena A-4 second stage that incorporates strapdown inertial guidance; and the TE-364-4 third stage. This new model of Thor/Agena can be available two years after request for launch, will cost about $7 million and be capable of injecting 4500 pounds into low earth orbit, or about 1600 pounds into a geo-synchronous transfer orbit.

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INTRODUCTION

The Delta and Thor/Agena launch vehicles offer scientific and applications satellite mission planners a broad spectrum in performance capabilities together with unprecedented mission flexibility. Depending on the mission, these two medium class launch vehicles can be configured on the new Universal Boattail (UBT) Thor booster in either two or three stages with thrust augmentation of the UBT ranging from three to nine strap-on solid propellant motors. Both vehicles incorporate strapdown inertial guidance systems that allow flexible mission programming by computer software changes rather than by hardware adjustments. Where vehicle performance exceeds the requirements of the primary mission, support and separation systems are qualified and flight proven for carrying secondary experiments or ejectable satellites on the Delta and Agena second stages with options for precise on-orbit altitude orientation for long durations. The Delta launch vehicle is available for launch from both the Eastern Test Range (ETR) in Florida and or the Western Test Range (WTR) in California; a Thor/Agena launch capability at either site can be made readily available.

The Delta Model 904 and the Thor/Agena Model 9A4 scientific and applications satellite launch vehicles are described for potential users, together with projections of future growth and launch costs.

I. DELTA

A. The Evolution of Delta

The evolution of the Delta launch vehicle, shown in Figure 1, reaches back fifteen years when, in 1955, the United States participated in the International Geophysical Year (IGY) and undertook the development of the Vanguard three-stage launch vehicle; in the same year the Air Force initiated the development of the Thor IRBM. With modifications, the Thor became the first stage of Delta; the Vanguard second stage propulsion system, evolved through the Able programs, became the Delta second stage propulsion system; and the Vanguard X-248 third stage solid propellant rocket motor was adapted as the third stage for Delta. The development and integration of these systems and the production of twelve (12) vehicles was started in early 1959 under prime contract to the Douglas Aircraft Company, now McDonnell-Douglas Astronautics Corporation (MDAC). The initial objective of the Delta program was to provide an interim space launch vehicle capability for the medium-class payloads until more sophisticated vehicles as Scout and Agena, then under development, could be brought to

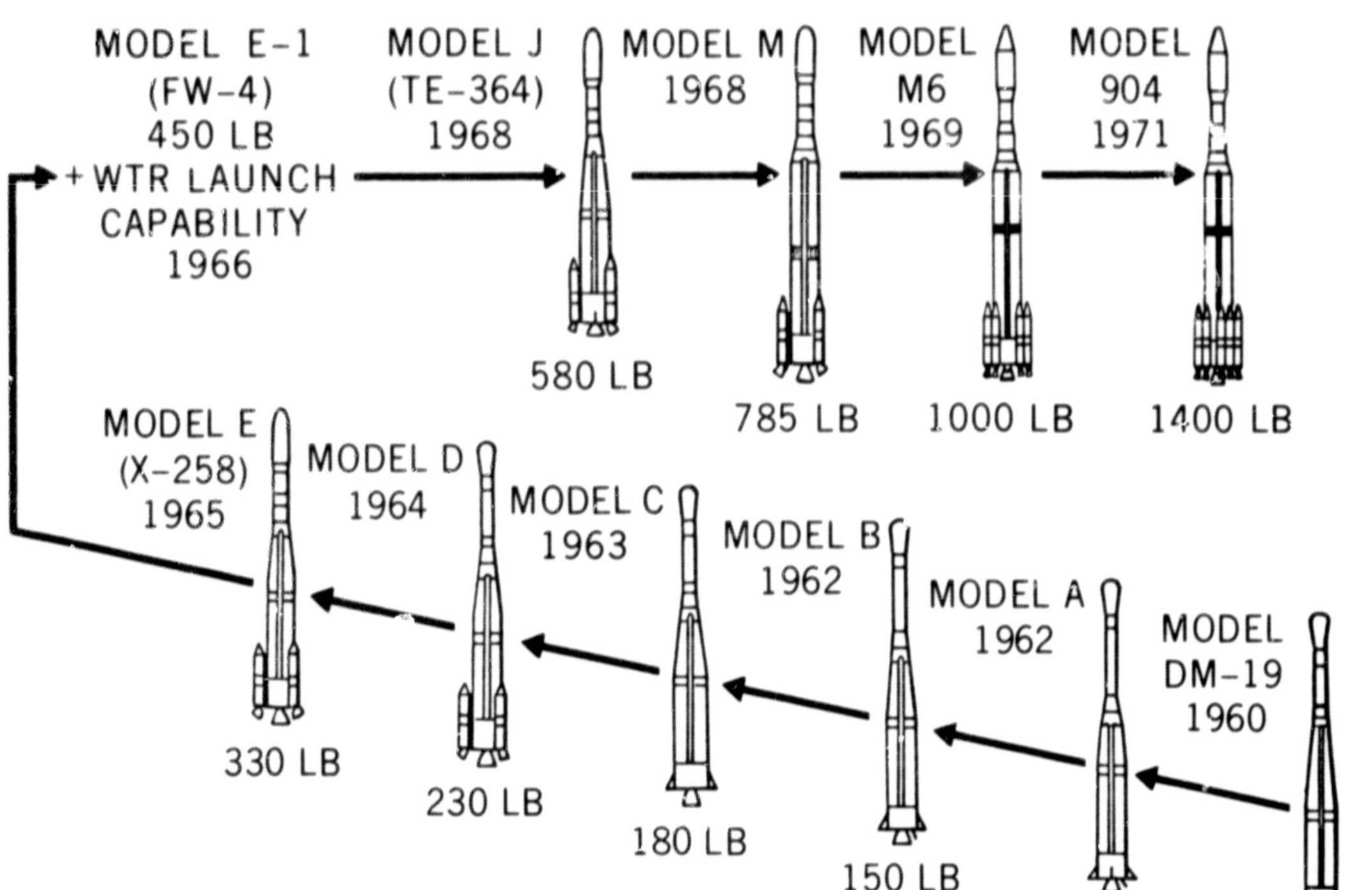


Figure 1. Delta Evolution

operational status. The development program spanned 18 months. In a little over two years, following the development period, eleven of the twelve vehicles were launched successfully carrying, among others, the first passive communications satellite, Echo I (August 1960), the cooperative NASA/United Kingdom Ariel I (April 1960), the TIROS II through VI series, the first Orbiting Solar Observatory, and the first private industry satellite, AT&T's Telstar I (July 1962). The total development cost, including the twelve vehicles (Model DM-19) and launch support, was approximately $43,000,000, compared to the $40,000,000 estimated at the outset of the program.

Before the development program was complete the number of missions planned for Delta outstripped the interim buy of twelve vehicles, so an order was placed for fourteen additional vehicles. This follow-on buy of Deltas (Models A and B) incorporated lengthened second stage propellant tanks, a higher energy second stage oxidizer, transistorized guidance electronics, and assiduous application of high-reliability semiconductors in flight critical circuits. This model of Delta carried NASA's first active communications satellite, Relay I (December 1962), and the first synchronous satellites, Syncom I and II (February and July 1963).

The next production order of Deltas (Models C and D) in 1963 brought the adaption of the USAF developed improved Thor booster with thrust augmentation provided by three strap-on solid propellant motors and the adaption of the Scout developed X-258 to replace the X-248 third stage motor. The first thrust augmented Delta (TAD) carried Syncom III (August 1964), the first equatorial synchronous communications satellite. The second TAD vehicle orbited the first commercial communications satellite, Comsat Corporation's Early Bird Satellite (April 1965).

Another order of Deltas in 1964 brought the development of the Improved Delta (Model E). The Improved Delta model adapted and extended the large diameter propellant tanks from the Able-Star stage, and thereby nearly doubled the propellant capacity of the previous Delta second stage. The larger diameter tanks in addition permitted adaption of the five foot diameter Nimbus fairing developed for the USAF Agena stage. Improved Delta also adapted the USAF developed FW-4 solid propellant motor to replace the X-258 third stage motor (Model E1). The first Improved Delta was launched November 1965 and among the missions carried on this model of Delta are the near polar Geophysical Orbiting Satellites, GEOS A and B; the heliocentric Pioneer series A through D; the low earth orbiting Biological Satellite, BIOS A through C; the synchronous communications satellites, Intelsat F1 through F4; the lunar orbiting Anchored Interplanetary Monitoring Probe, A-IMP A and B; the sun-synchronous ESSA 2 through 6; the High Eccentric Orbiting Satellite, HEOS developed by the European Space Research Organization and the Canadian International Satellite for Ionospheric Studies, ISIS.

In 1966 Delta undertook to adapt the Surveyor spacecraft solid propellant retromotor as a new third stage. The spherical case was modified to mate to a spin-table assembly and the motor, designated TE-364-3, was requalified for the Delta spinning environment. The first Delta using this third stage motor, Delta Model J, was launched in July 1968 and carried the Radio Explorer, RAE-A spacecraft.

At about the same time Delta initiated the adaption of the TE-364-3 motor, the USAF undertook the uprating of the Thor booster by lengthening the liquid oxygen and RP-1 fuel tanks and converting the fuel tank to a constant 8 foot diameter. This Long Tank Thor carries about 47 percent more propellants than previous models. In September 1968, Delta launched its first Long Tank Thor with the Improved Delta second stage and TE-364-3 third stage. The Delta Model M carries, among others, the Intelsat III series and the British Skynet and NATO communications satellites.

In early 1968, Delta started a redesign and retrofit of the Long Tank Thor engine section to permit the addition of a second set of three thrust augmentation solid motors. The first Delta Model M6 with six solid motors was launched from the Western Test Range in January 1970 and carry the NASA TIROS Operational Satellite, TOS-M into a 800 n. mi. circular sun-synchronous orbit. The two remaining Delta Model M6 vehicles are to be used to carry the Interplanetary Monitoring Probe I and the ITOS-A spacecraft.

To date, Delta is launching over fifty percent of NASA's unmanned spacecrafts each year and has been selected for the use of private industry and foreign governments. The reliability and cost effective history of Delta is, in a large part, attributable to the technical approach taken at the outset of the program and still adhered to today. This approach is to use current technology and flight proven components wherever possible from other space programs. The resultant vehicle is normally heavy, but cheap and has a high probability of performing repeatedly and reliably from the outset. Delta has never considered it necessary to have a pre-operational or development flight test launch for any of the ten major changes made to the vehicle. And with the exception of the first Delta launch in 1960, there has never been a failure of the first flight article on its maiden launch. The criteria for evaluating improvements to Delta is that they must meet the mission requirements at the lowest possible cost and risk without compromise of Delta reliability record—currently 74 successes out of 80 launches for a cumulative success rate of 92%. For this reason Delta has wherever possible, adapted flight proven components from other programs. The current evolutionary uprating of Delta is consistent with this past pattern of change.

To keep pace with the growing launch capability requirements of scientific and applications satellites both domestic and foreign, the Delta launch vehicle is

being uprated in performance capability, in guidance accuracy, and modernized to enhance overall systems reliability. This new Delta (Model 904) with nine strap-on solid motors is shown in Figure 2 and is composed of the Universal Boattail (UBT) Long Tank Thor designed to accept thrust augmentation from combinations of 3 to 9 strap-on solid motors; the Delta second stage uprated with a new engine, Aerozine - $50/N_2O_4$ propellants and a strapdown inertial guidance system; and the Thiokol TE-364-4 solid propellant third stage motor. This Delta is described together with its performance, flight environment, and launch costs. The Delta Model 904 is to be available in late 1971 and is now scheduled to carry the Planetary Explorer series, the Interplanetary Monitoring Probes J and K, Earth Resources Satellite series and the Synchronous Meteorological Satellites.

B. Vehicle Description

The three stage Delta vehicle, Model 904, shown in Figure 2 stands 106 feet and weighs 261,000 pounds at lift-off. The vehicle is designed for ascent through

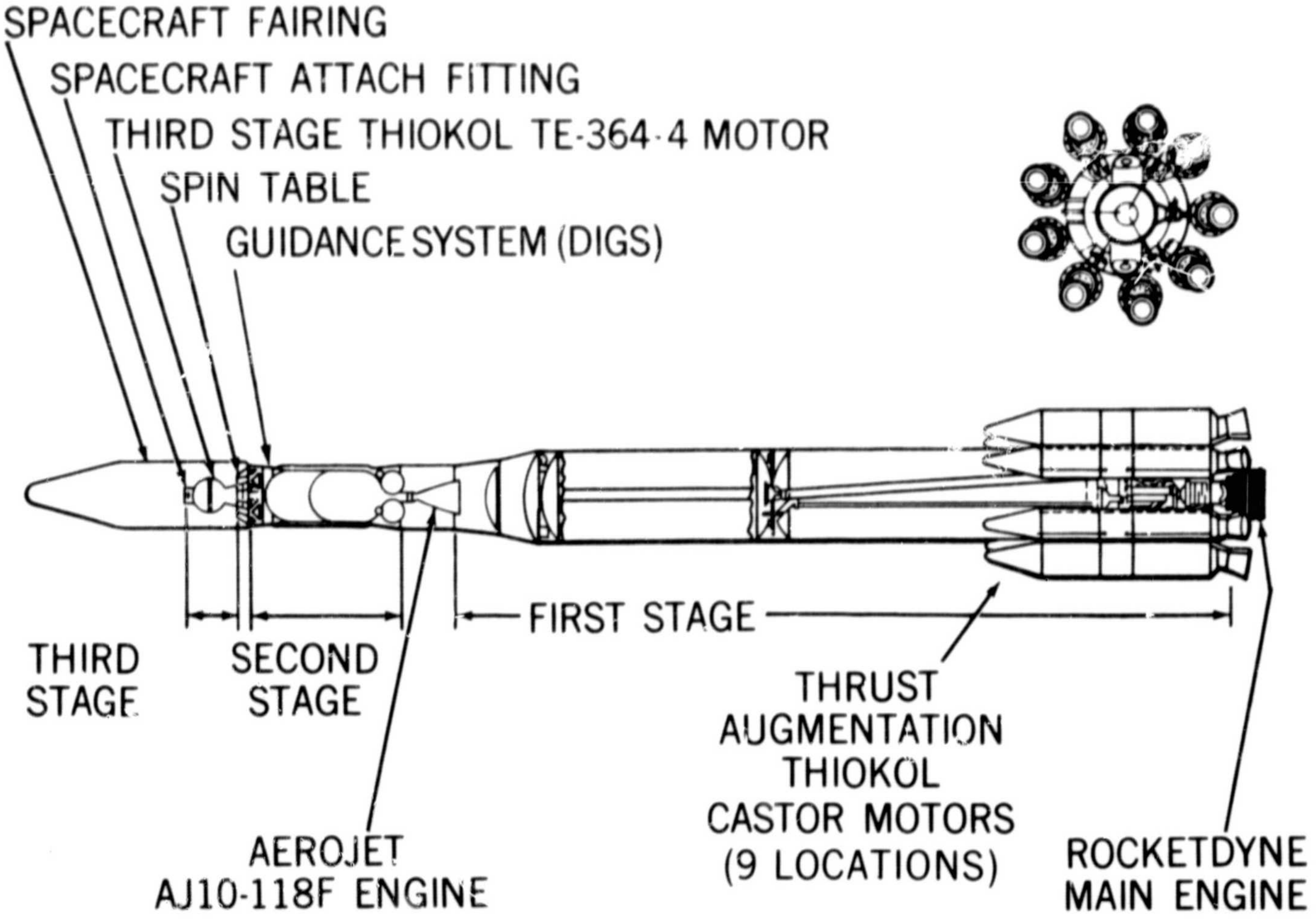


Figure 2. Delta Model 904

95% ETR and WTR upper atmosphere annual wind profiles, lift-off in 40 knot ground winds, and hold on the launch pad for several hours in readiness for launch windows only seconds wide.

The first stage liquid propellant core is the reliable Long Tank Thor produced by MDAC. The core is 8 feet in diameter, 60 feet long, carries 147,000 pounds of RP-1 and liquid oxygen propellants. It is powered by a turbopump fed Rocketdyne main engine that develops 175,000 pounds thrust at lift-off. The core burns to propellant depletion about 220 seconds after lift-off (T+ 220) at an altitude of 60 to 70 nautical miles (n.mi.). Thrust augmentation solid propellant motors attach at the base of the first stage core to the UBT engine section structure. The UBT is structurally designed and thermally insulated to carry up to 9 Thiokol Castor II solid motors (TX-354-5) or 3 Algol class of solid motors with six Castor II's. Use of the Algol class of solid motors is not approved at this time. The UBT uses the same solid motor attach and separation hardware as on all previous Thrust Augmented Thors. The simple, straightforward structural, pneumatic and electrical modifications to the present engine section obviates the need for extensive testing or a preoperational flight test.

Normally, the thrust augmentation motors are build-up in sets of three. Up to six motors can be ignited on the pad and the remainder no sooner than 38 seconds after lift-off in order to hold the vehicle acceleration induced loads on the propellant tank bottoms within allowable limits. The Castor II motors each develop 33,000 pounds thrust at ignition, burn for 40 seconds and are jettisoned from the core in sets of 3 at 5 seconds intervals starting at T+ 85. This time is dictated by considerations of combined dynamic pressure angle-of-attack loadings on the jettison mechanism and a Range Safety requirement for an offshore impact of the expended motors. Jettison is effected by firing an explosive bolt holding a clamped ball-socket joint. Acceleration of the core plus aerodynamic drag on the motors eject the cases away from the vehicle as shown in Figure 3.

During powered flight pitch and yaw steering is exerted by gimballing the core main engine. Roll control is effected by differentially gimballing a pair of small outboard vernier engines. Subsequent to main engine shut-down the verniers continue to operate for about 6 seconds, damping shut-down transients and stabilizing the vehicle for staging of the second stage.

Previous to the Delta, Model 904, guidance steering commands were accomplished with radio guidance which is constrained by radar look angles and must be augmented by a velocity-sensing system or downrange tracking for guidance over-the-horizon capability. These limitations of the radio guidance system are eliminated with the new Delta Inertial Guidance System (DIGS) that is located in the second stage and provides guidance and control for the total vehicle from

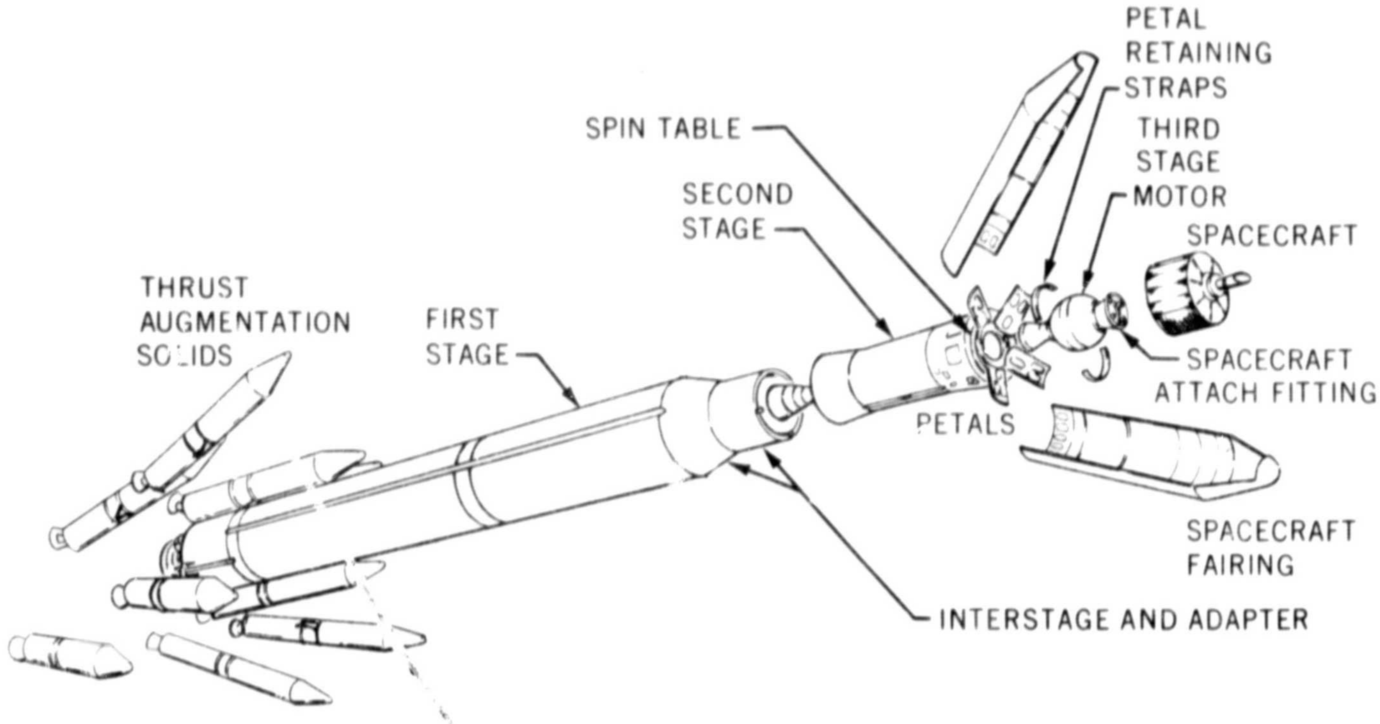


Figure 3. Delta Staging Schematic

lift-off through attitude orientation and ignition of the spin stabilized third stage solid propellant motor. This strap-down system is composed of a digital computer developed by Teledyne for the Advance Centaur vehicle and an inertial measurement unit (IMU) developed by Hamilton Standard for the Apollo Lunar Excursion Module Abort Sensing Assembly. The 4096 word memory computer performs the navigation, guidance, steering, controls systems stability and shaping, and initiates discretes for both first and second stages. It directs the vehicle through a pre-programmed trajectory navigating on IMU velocity data to determine present position and velocity, which it then predicts forward along a nominal trajectory to determine the final position and velocity at injection. The predicted final terminal state is compared to the desired terminal state to derive the required vehicle steering commands and engine shut-down time to reach the desired terminal injection state. All guidance functions are programmed into the vehicle computer with launch pad computer software rather than hardware adjustments. This permits maximum mission flexibility for the user.

The interstage section between the first and second stages is provided with a spring separation assembly. Eight seconds after first stage main engine shutdown explosive bolts that attach the two stages are fired and the second stage is spring separated from the first stage. Three seconds later the second stage engine is started.

The Delta second stage is 17 feet long, approximately 5 feet in diameter and weighs 12,000 pounds at ignition. The Aerojet engine originally developed for

the Titan III Transtage vehicle and now adapted by Delta, is a pressure fed, ablative and radiation cooled engine that developes 9460 pounds thrust at an up-rated 125 pounds per square inch chamber pressure and operates for about 345 seconds on Aerozine-50 and N_2O_4 storable propellants. The propellant tanks are cylindrical with a hemispherical internal common bulkhead between the fuel and oxidizer tank. The system is pressurized from lift-off to strengthen the structure and suppress oxidizer boiling. The engine, designated AJ 10-118F, is capable of multiple restarts and is started by actuation of a single bi-propellant value.

During the second stage powered flight, pitch and yaw steering is provided by gimballing the engine and roll is controlled by cold nitrogen gas jets. Cold nitrogen gas jets control the vehicle in all axes during coast and provides propellant settling ullage thrust for restarting the engine. The control system electrical power and nitrogen gas supply is capable of maintaining second stage attitude for a little over two hours. For long second stage coast periods before third stage spin-up and separation, the second stage may be reoriented with respect to the sun or the vehicle placed in a slow yawing or pitching tumble to alleviate assymetric solar heating of the spacecraft.

Peripheral second stage systems include a "C" band tracking beacon, a PDM/PCM/FM/FM 45 × 20 "S" band telemetry system, dual command destruct receivers and associated power supplies.

On several Delta missions where the second stage was orbited and the vehicle performance exceeded the requirements of the primary mission, the second stage was used as a platform for placing secondary satellites into orbit. Table 1 summarizes the secondary satellites that have been carried on the Delta second stage and ejected into orbit after either the primary spacecraft or the third stage with the primary spacecraft had been separated from the second stage. Included also in Table 1 are the secondary experiments and satellites currently under active consideration for piggyback flights on Delta.

Secondary experiments or satellites can either remain on-board the second stage or be ejected. Support and separation systems have been qualified and flight proven for ejecting satellites. For experiments that remain on-board, an orbiting Delta second stage secondary experiment recently demonstrated the feasibility of providing on-board experiments with power, data and command RF links, passive thermal control and earth-oriented attitude pointing for long duration.

In August 1969, Delta launched a Packaged Attitude Control (PAC) system experiment, utilizing the orbiting expended Delta second stage which was used to inject the Orbiting Solar Observatory (OSO-G) into orbit. PAC demonstrated the feasibility of making platforms for earth-oriented experiments out of an otherwise

Table 1

Delta Secondary Experiments/Satellites

Secondary Experiment/ Satellite	Primary Mission	Date	Experiment/ Satellite Wt. (lbs)	Orbit
PAST MISSIONS				
TETR-A	PIONEER-C	12/67	55	160 × 260 n.mi. × 28.5°
TETR-B	PIONEER-D	11/68	55	240 × 500 n.mi. × 28.5°
PAC	OSO-G	8/69	265	300 n.mi. circ. × 33°
OSCAR V	TIROS M	1/70	40	790 n.mi. circ. × 101.6°
MISSIONS UNDER CONSIDERATION				
CEP	ITOS A	1970	11	790 n.mi. circ. × 101.6°
TETR-D	OSO-H	1971	66	300 n.mi. circ. × 33°
OSCAR VI		1971	53	
INTASAT	SUN SYNCH MISSION	1972-73	77	300 n.mi. circ.
HSS	NIMBUS E	1972	88	600 n.mi. circ. × 100°

expended stage by use of a new low-cost attitude control system. Using gravity gradient torqueing via the long thin stage, the Delta PAC is a three-axis attitude control system that aligns the second stage roll axis to the earth geocenter and the pitch axis normal to the orbit plane. The PAC system includes a power

supply (batteries, solar panels and electronics package), a telemetry subsystem, a command subsystem, a magnetic moment assembly, solar aspect indicators, a passive thermal system utilizing heat pipes, and the PAC attitude control system.

Active pitch control is provided by a reaction wheel driven in response to pitch information provided by a horizon scanner. The gyroscopic action of the same reaction wheel in conjunction with gravity gradient torques provide the mechanism for roll/yaw control. Damping of roll/yaw motion is provided by gimballing the wheel and coupling it to the stage through an eddy current damper and a very weak spring suspension. The motor in the reaction wheel scanner provides gyroscopic action, provides a reaction torque by accelerating or decelerating when a pitch error is sensed, and rotates the optics in an infra-red optical system, producing a cone scanning action, which locates the earth's horizons, with respect to PAC. In addition, this motor has magnetic pickoffs which are used to furnish both speed information and vertical reference data. The motor provides all this capability with the use of only a few watts of power.

Although the system is designed to require no attention from the ground, a number of options are available by ground command which modify the control laws. The stage can be flown right-side-up, upside-down, frontwards, or backwards. The nominal speed of the wheel is adjustable, as is the amount of tachometer feedback. In addition, the scanner null can be electronically adjusted to correspond with the gravity gradient null. All of these control law modifications, which are available by ground command, have been verified in orbit. Performance of the control system in orbit, to date, has been satisfactory. Attitude pointing of sensors of experiments has been maintained within a ±2 degree accuracy.

Planned Delta missions with trajectories that place the second stage into orbit are given in Table 2 together with the current excess performance capability that can be used to carry secondary experiments or satellities. Some missons show no excess capability; however, on these missions, an excess capability can be made available by additional thrust augmentation solid motors on the first stage. The excess performance shown are, of course, subject to changes as the primary mission spacecraft weight or orbital parameters change.

For this reason, experimenters who desire to utilize the Delta piggyback capabilities must work closely with the Delta/Agena Project Office to ensure mission compatibility and performance availability.

Possible orbits for secondary experiments and satellites are not necessarily constrained to those shown for the primary spacecraft in Table 2 since the second stage can be restarted and injected into a new orbit after the primary spacecraft is deployed.

Table 2

Future Delta Missions with Orbiting Second Stage

Year	Mission	Second Stage Orbit	Current Excess Capability
1971	OSO-H	300 NM. Circular i = 33°	—
1971	ITOS-D	790 NM. Circular i = 101.56°	50 lb
1972	HEOS-A2	216 NM. Circular i = 90°	—
1972	TD-1	300 NM. Circular i = 97.4°	—
1972	ERTS-A	500 NM. Circular i = 99.16°	—
1972	NIMBUS-E	600 NM. Circular i = 110°	<25 lb
1972	GEOS-C	500 NM. × 650 NM. i = 20°	250 lb
1972	ITOS-C	790 NM. Circular i = 101.56°	390 lb
1972	SMS-A	100 NM. Circular i = 28.5°	70 lb
1972	RAE-B	100 NM. Circular i = 28.5°	480 lb
1973	ERTS-B	500 NM. Circular i = 99.16°	—
1973	NIMBUS-F	600 NM. Circular i = 100°	<25 lb
1973	OSO-I	300 NM. Circular i = 33°	500 lb
1973	SMS-B	100 NM. Circular i = 28.5°	70 lb
1973	AE-C	81 NM. × 2160 NM. i = 80°	250 lb
1974	OSO-J	300 NM. Circular i = 33°	500 lb
1974	AE-D	81 NM. × 2160 NM. i = 105°	250 lb
1975	OSO-K	300 NM. Circular i = 33°	500 lb

The third stage assembly consists of a spin table, the Thiokol TE-364-4 solid propellant motor, spacecraft attach fitting, spacecraft and the spacecraft fairing. The spin table shown in Figure 4 consists of a bearing support structure and a conical third stage motor pedestal truss that is divided into four petals hinged at the base and clamped to the equator of the third stage motor by a retaining strap. The retaining strap is held in tension by two explosive bolts that are fired two seconds after the motor and spacecraft are spun up and the 15 second time delay squib that ignites the TE-364-4 motor is started. The released petals fly outward under centrifugal force, releasing the third stage from the spin table (Figure 3). At the same instant the second stage is backed away from the free spinning third stage by venting residual pressurant (helium) overboard through two retrojets.

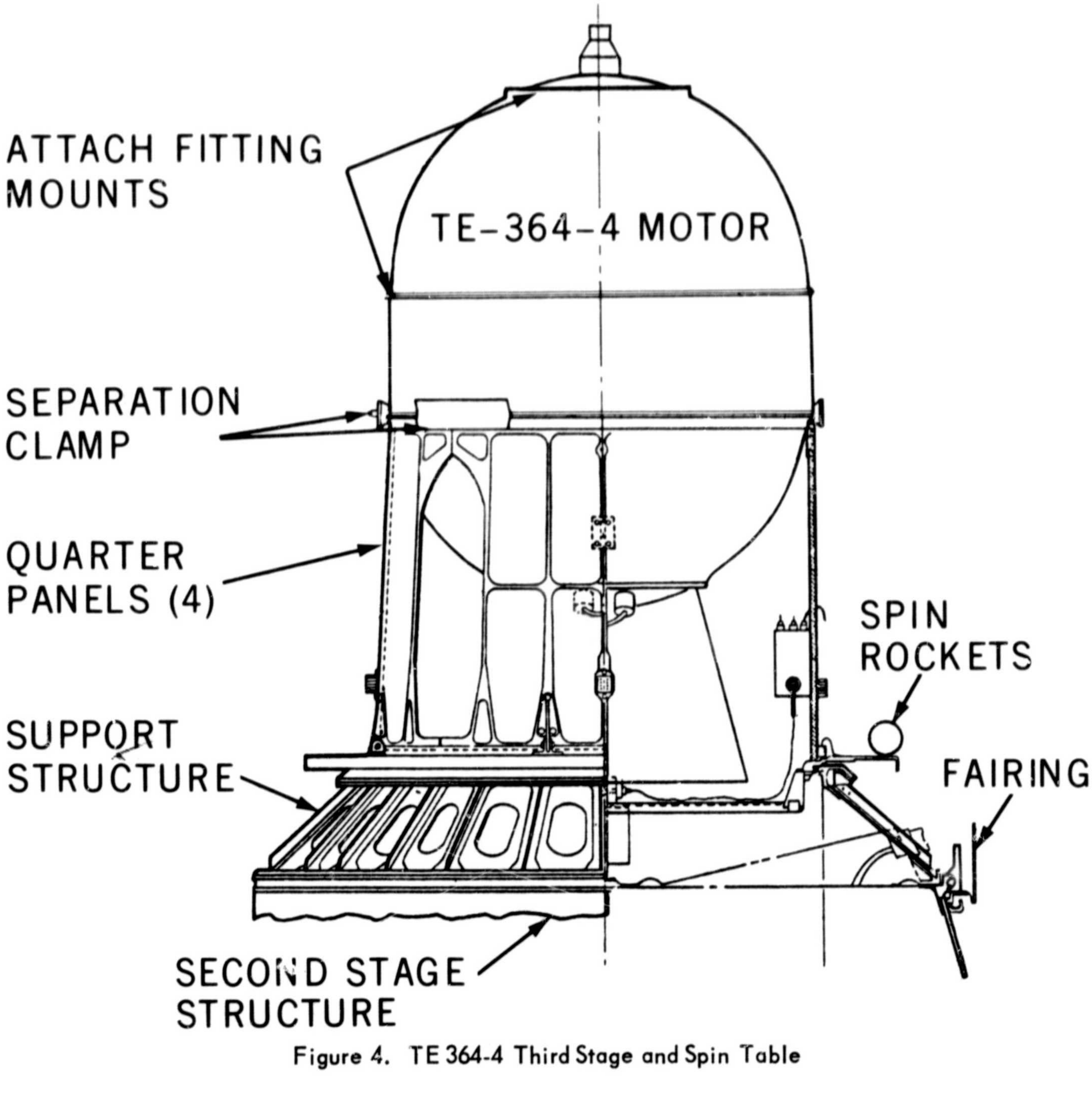


Figure 4. TE 364-4 Third Stage and Spin Table

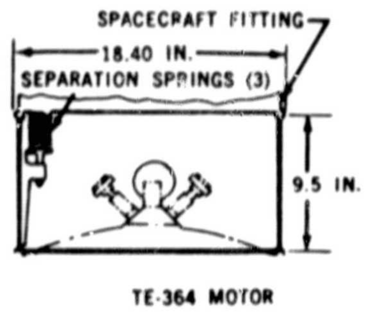
Approximately thirteen seconds later the third stage motor is ignited by the time delay squib. The TE-364-4 motor is essentially identical to the -3 model that is in current use except that a 14 inch cylindrical section is added between the two hemispherical halves of the case. The propellant weight is increased to 2300 pounds from 1440 pounds, it burns for 44 seconds and develops an average thrust of 15,000 pounds.

Torque to the spin table is imparted by combinations of small solid propellant rocket motors, which provide spin rates from 30 to 100 rps (±10 percent) for spacecraft roll moments of inertia ranging from 20 to 170 slug-feet squared. A lower limit of approximation 30 rpm is dictated by minimum dynamic stability of the third stage/spacecraft during third stage motor burning. If less than 30 rpm is desired the effect upon orbit injection errors must be carefully assessed. The anticipated maximum spin rate users would desire was 100 rpm, consequently the third stage motor is qualified only up to this spin rate.

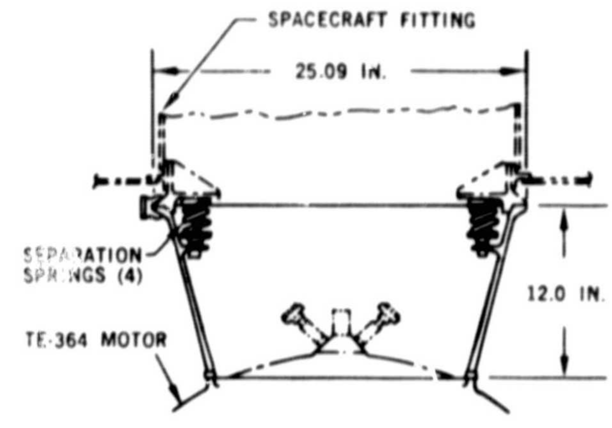
For those spacecraft that require spin stabilization but mission performance does not require use of a third stage, a spacecraft can be spun either by use of the spin table or by placing the combined second stage/spacecraft in a controlled spin with the second stage roll attitude control jets. This technique, which was used to spin up the TOS-M spacecraft (Delta 76) eliminates the spin table for missions requiring spacecraft spin rates up to 20 rpm.

The spacecraft is clamped to the attach fitting by a circular retaining strap assembly that releases by firing two explosive bolt cutters subsequent to third stage motor burn-out. Separation from the expended third stage is then effected by a separation spring, or springs, which provides the spacecraft with a relative separation velocity of 6 to 8 fps with respect to the expended third stage motor. Although peculiar spacecraft requirements may dictate the design of a special spacecraft attach fitting, a number of standard Delta fittings are available. These are shown in Figure 5. These fittings use either a small rocket or yo weight system to tumble the expended third stage motor after spacecraft separation to preclude possible motor outgassing from accelerating it into the spacecraft. Also available is a yo-yo weight despin system which can despin the third stage and spacecraft combination prior to spacecraft separation. Attach fittings include timer assemblies, battery and delay squib switches. The timers are initiated by the second stage computer and run on mechanical energy until reaching a predetermined time to fire the spacecraft separation clampband bolt cutters and a pair of squib switches. Two seconds later the squib switches initiate a small rocket or yo weight to tumble the expended third stage motor.

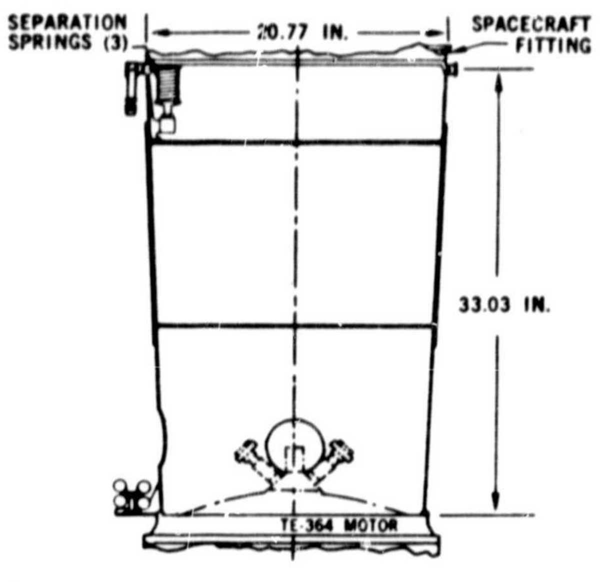
For users requiring real time third stage motor performance, environmental or velocity increment information an "S" band telemetry system and a "C" band



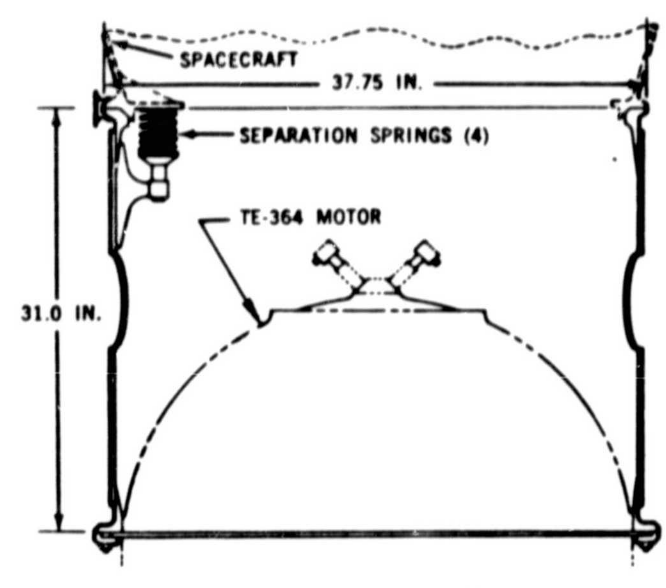
18 × 9 INCH CYLINDRICAL ATTACH FITTING
WT: 24 LB



25 × 12 INCH CONICAL ATTACH FITTING
WT: 30 LB



20 × 30 CONICAL ATTACH FITTING
WT: 24–30 LB



37 × 31 INCH CYLINDRICAL ATTACH FITTING
WT: 54 LB

Figure 5. TE 364 Motor/Payload Attach Fittings

tracking beacon are developed and flight proven. These are carried on either the spacecraft attach fitting or on the third stage motor as optional equipment.

The spacecraft fairing is fiberglass, and constructed in two-half-shells that are brought up around the spacecraft laterally and clamped together by three strap assemblies that are released in flight by explosive bolts. Spring cartridges thrust the half-shells laterally and pivots at the base of the fairing cause the shells to rotate rearwards and clear the vehicle (Figure 3). Normally, the fairing is jettisoned within 5 to 20 seconds after second stage start. Fairing jettison time is dictated by the free molecular heating rate that can be tolerated by the spacecraft. Normally, the heating rate is held below 0.1 BTU/Ft^2-sec. or about

equivalent to the solar heating rate to the spacecraft. Aerodynamic heating of the fairing is controlled by application of ablative materials to hold the fairing internal temperature to below 450°F. This precludes any possibly of spacecraft contamination from outgassing of the fiberglass phenolic.

Access ports through the fairing are provided at the locations that meet the needs of the vehicle user. The available fairing internal envelop is shown in Figure 6.

C. Flight Sequence and Performance

The Delta flight profile and sequence of events for a three stage geo-synchronous transfer mission having a perigee altitude of 100 n.mi., an apogee altitude of 19,400 n.mi. and an inclination of 28.5 degrees is shown in Figure 7. The vehicle is launched from ETR on an azimuth of 95 degrees. The third stage assembly is placed into a 100 n.mi. parking orbit and coasts to a point just short of the Equator

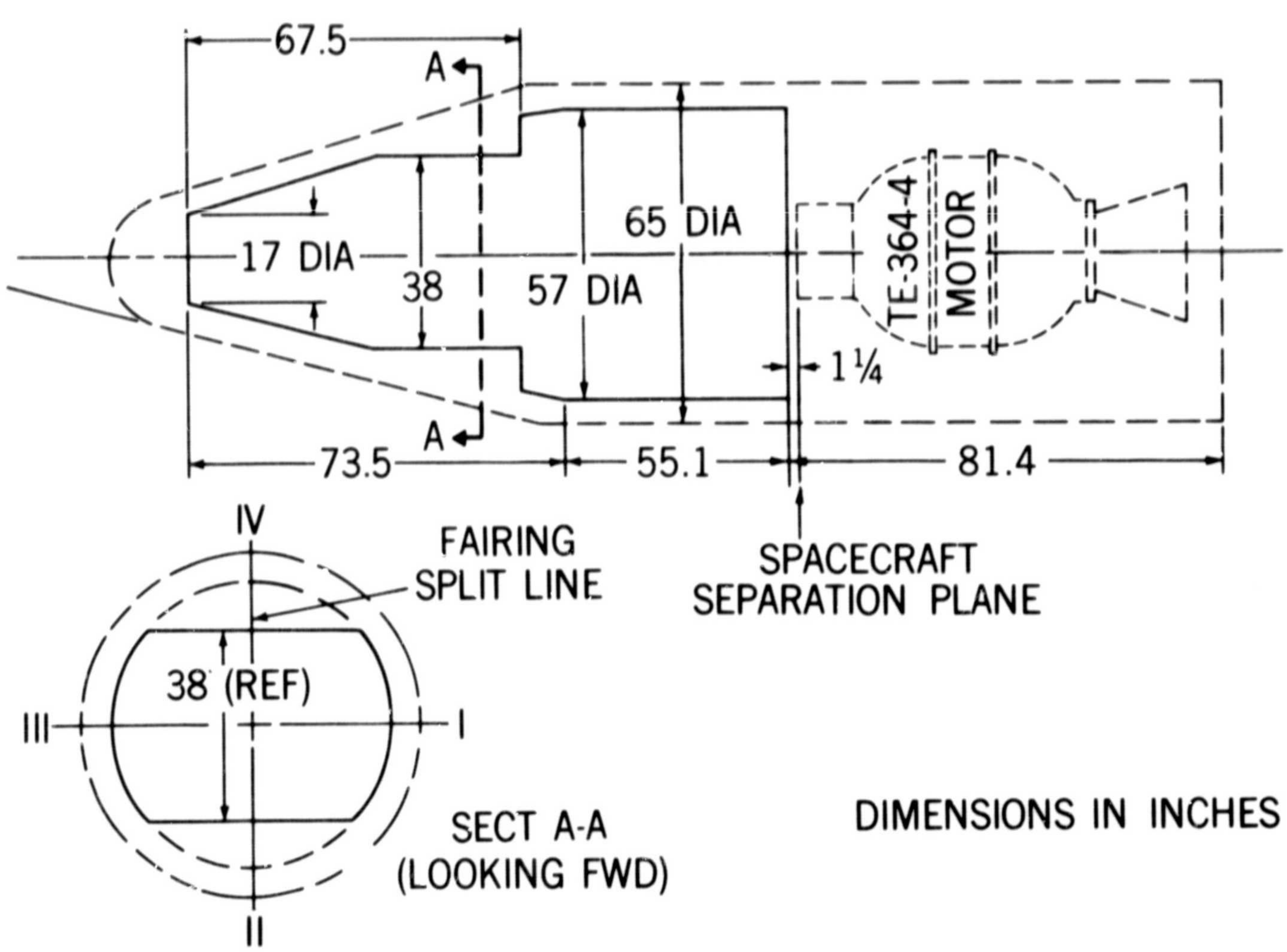


Figure 6. Delta/Agena Spacecraft Fairing Envelope

NO.	TIME (SEC)	EVENT	NO.	TIME (SEC)	EVENT
0	0	LIFT-OFF			
1	85	JETTISON SOLID MOTORS	5	552	SECOND STAGE CUT-OFF
2	218	FIRST STAGE BURNOUT	6	1443	THIRD STAGE IGNITION
3	222	SECOND STAGE IGNITION	7	1487	THIRD STAGE BURN-OUT
4	228	FAIRING JETTISON	8	1592	SPACECRAFT/THIRD STAGE SEPARATION

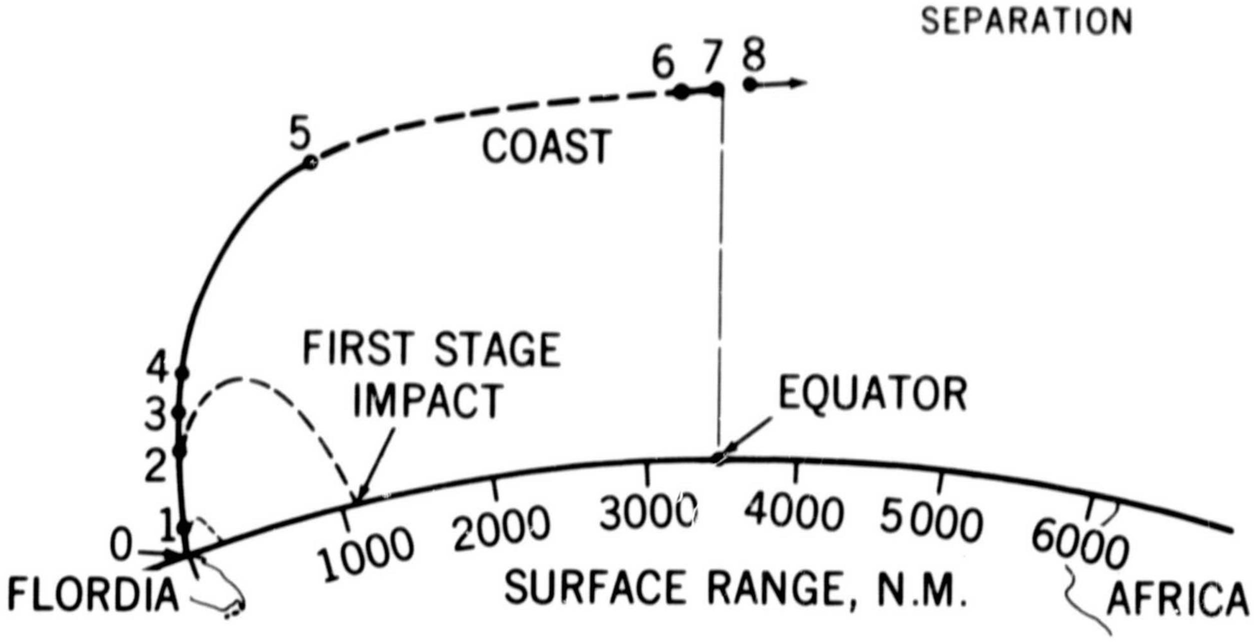


Figure 7. Delta Flight Sequence of Events for a Geo-Synchronous Transfer Mission

where third stage spin-up, separation and ignition occur. The third stage burns out directly over the Equator at an altitude of 100 n.mi., an inertial flight path angle of zero degrees, and with sufficient velocity to coast the spacecraft to an altitude of 19,400 n.mi. on the opposite side of the Earth so that the line of apsides lies in the equatorial plane to permit the spacecraft apogee motor to rotate the transfer orbital plane into the equatorial plane as part of the circularization maneuver.

Payload weight versus characteristic inertial velocity for Delta from the Eastern Test Range in Florida and the Western Test Range in California is shown in Figures 8 and 9. The performance capability for a number of scientific and applications missions carried on Delta is summarized in Table 3. These Delta performance capabilities are the useful load that can be carried above the last powered stage and thus includes the spacecraft weight and its attach fittings hardware weight. The first number in the Delta model designation (3, 6, or 9) indicates the number of thrust augmentation solid motors; the second digit (0) Delta second stage with DIGS and the AJ 10-118F engine; the third digit (4) denotes the -4 version of the TE-364 third stage motor.

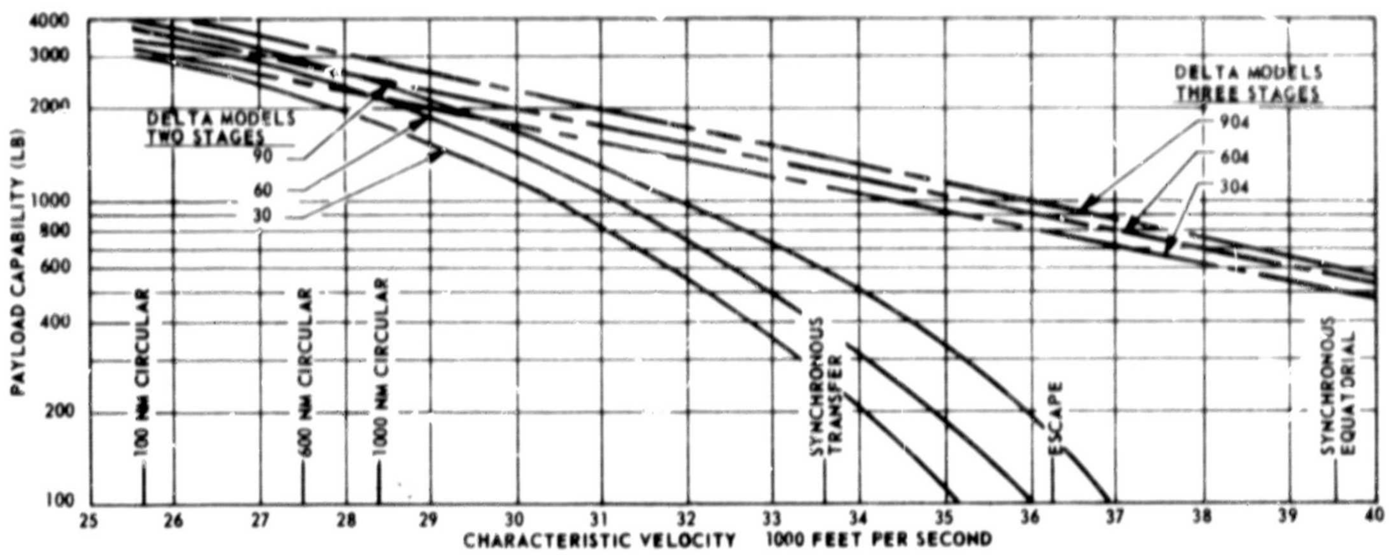


Figure 8. Delta Payload Capability – Eastern Test Range

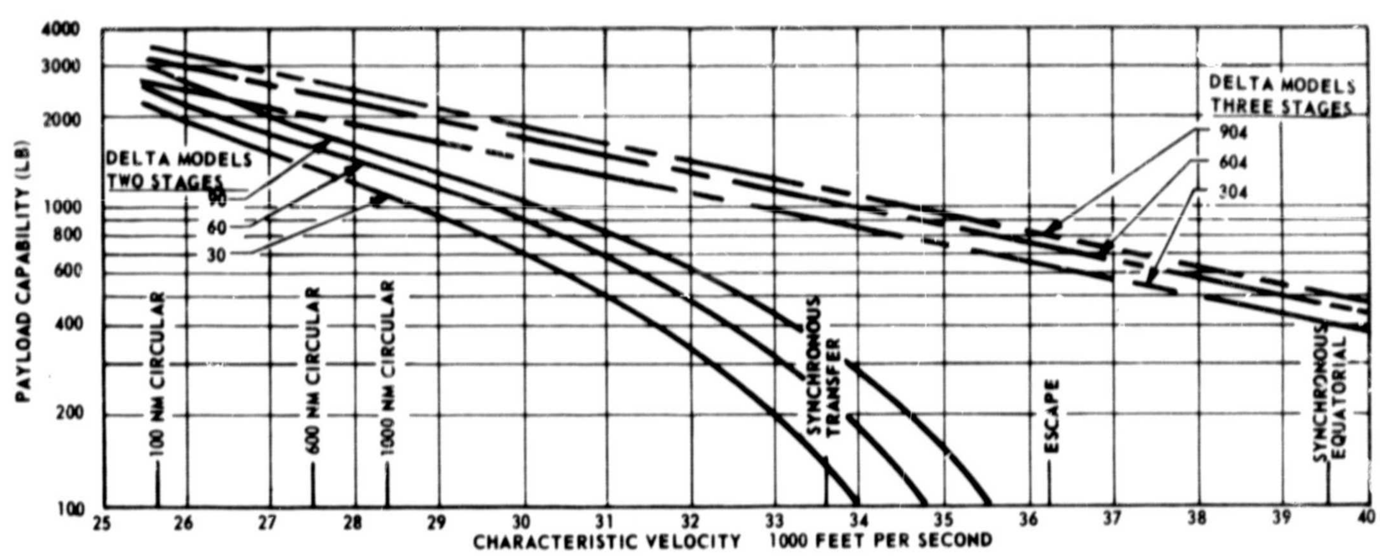


FIGURE 9. DELTA PAYLOAD CAPABILITY – WESTERN TEST RANGE

Figure 9. Delta Payload Capability – Western Test Range

The injection accuracy of Delta is strongly dependent on the trajectory profile. The single largest injection error source is the unguided, spin-stabilized third stage. Nearly two thirds of the errors in injection velocity and attitude are caused by dispersions in the motor total impulse and lateral tip-off impulses applied during separation from the second stage and at motor ignition. Typically, the 99 percent probability dispersions for a nominal 100 n.mi. by 19,400 n.mi. geo-synchronous transfer trajectory is shown in Table IV.

Table 3

Delta Performance Capabilities

Missions	Flight Mode	Delta Performance Capability-Pounds Model Number		
		304	604	904
Biosatellite 200 N.M. Circular Incl. = 28 Degrees	Two Stage, Restart 100 × 200 N.M. Hohmann Transfer.	2900	3500	3800
Earth Resources 500 N.M. Circular Sun-Synchronous Incl. = 99 Degrees	Two Stage, Restart 100 × 500 N.M. Hohmann Transfer.	1600	2000	2250
Improved TIROS Operational Satellite (ITOS) 800 N.M. Circular Sun-Synchronous Inc. = 102 Degrees	Two Stage, Restart 100 × 800 N.M. Hohmann Transfer.	1300	1600	1850
NATO-A Synch. Transfer 100 × 19,400 N.M. Incl. = 28.5 Degrees	Three Stage, Direct Ascent. Second Stage Placed in 100 N.M. Parking Orbit.	1075	1250	1400
Planetary Explorer Venus Type II $C_3 = 8.231$ KM ^{2}/SEC. ^{2}	Three Stage, Direct Ascent. Second Stage Placed in 100 N.M. Parking Orbit.	650	750	825

Though these dispersions are moderately large, it should be remembered that a synchronous communications satellite must carry a propulsion system for station keeping and hence the penalty paid in additional propellant to trim out injection errors is quite small.

Table 4

Geo-Synchronous Transfer Orbit Dispersion

Parameter	Nominal	99% Probable Dispersion
Apogee Altitude, n.mi.	19,400	±600
Perigee Altitude, n.mi.	100	±8
Orbit Period, minutes	663	±22
Orbit Eccentricity	0.73	±0.006
Orbit Inclination, degrees	28.5	±0.55

D. Flight Environment

The environment imposed on the spacecraft by the vehicle is estimated from previous flight measurements. A summary of the expected environment for both the two and three stage Delta vehicle is provided in Table 5 for use in preliminary studies by spacecraft mission planners.

At liftoff the spacecraft is subjected to both lateral and longitudinal sinusoidal vibration that load the spacecraft structure dynamically. At the time the three stage Delta lifts off the launch pins and the umbilicals are simultaneously retracted, a 1000 pound spacecraft can experience a maximum of ±1.5g, zero-to-peak (O-P), in the vehicle lateral modal frequencies, which range from 5 to 14 Hz. Superimposed at this time is a ±1.5g (O-P) longitudinal 13 Hz oscillation. These combined liftoff oscillations typically last for two to five seconds with the peak acceleration lasting one to two cycles. During the last twenty seconds of first stage flight, the Thor exhibits a 20 Hz "pogo" longitudinal oscillation that builds up to ±4.5g (O-P) at the time the steady stage longitudinal acceleration has reached about 6.8g. The maximum first stage steady stage acceleration of 7.7g's is the highest imposed by the two stage Delta. For three stage Delta, the maximum steady stage acceleration is dictated by the TE-364-4 third stage and reaches 23g's for a 500 pound spacecraft; of 10g's for a 1500 pound spacecraft.

Random vibration measured at the third stage attach fitting and spacecraft show power spectrum densities between 0.001 and $0.06g^2/Hz$ from 20 Hz to 2000 Hz in both lateral and longitudinal axes. The principal source of random vibration is boundary layer turbulance over the fairing and the reverse slope of the second stage guidance compartment that excites the structure and feeds up through the third stage assembly to the base of the spacecraft. Acoustical excitation also contributes to the random levels experienced.

Table 5

Delta Critical Flight Environment

Excitation	Flight Event	Duration	Two Stage		Three Stage	
		Seconds	Frequency, Hz	Level	Frequency, Hz	Level
Sinusoidal Vibration Thrust Axis	Lift Off T + 210 sec.	2 to 5 5 to 7	5-17 17-23 23-100	1.5g (O-P) 4.0 1.5	10-17 17-23 23-100	1.5g (O-P) 4.5 1.5
Lateral Axis	Lift Off	2 to 5	5-14 14-100	1.3* 1.0	5-14 14-100	1.5* 1.0
Random Vibration Three Axis	Transonic & Max. Q	10 to 15	20-300 300-1000 1000-2000	+ 4db/octave $0.06g^2/Hz$ -6db/octave	20-300 300-2000	+ 3db/octave $0.02g^2/Hz$
Shock	Spacecraft Separation	0.001	1600g at 0.8 milliseconds Terminal Peak Saw Tooth		1400g at 0.3 milliseconds Terminal Peak Saw Tooth	
Steady State Acceleration	First Stage Burnout			7.7g	6.8	
	Third Stage Burnout				23.5g for 500 lbs. spacecraft 10g for 1500 lbs. spacecraft	
Acoustic	Lift Off and Transonic	10 to 15	142db 37 to 9600 Hz (peak level 800 to 1000 Hz)			

*For spacecraft weight of 1000 pounds. For lighter spacecraft, levels are higher.

At liftoff and transonic the overall accoustical level inside the fairing is approximately 142 db (referenced to 0.0002 dynes/cm^2) from 37.5 to 9600 Hz. These levels are present for about 10 seconds at lift-off and again for about 15 seconds at transonic.

Shocks occur at main engine start, thrust augmentation solid motors ignition and jettison, staging, fairing jettison, and spacecraft separation from the expended third stage. For three stage Delta, cutting the bolts to separate the spacecraft from the expended third stage imposes the most severe shock spectrum on the spacecraft. The third stage motor and spin table assembly act to absorb the high frequency excitation from other sources. Cutting the separation bolts results in an estimated shock spectrum equivalent to one-third millisecond, 1400g terminal peak sawtooth input.

E. Cost

The projected cost of Delta Model 904 reimbursable launches in 1972 from ETR is about $5-1/2 million dollars. This includes hardware, the launch services, trajectory software, spacecraft integration, launch support services, and NASA administrative charges. This cost does not include charges made by the U. S. Air Force for Range use that would include tracking, data acquistion, technical operations, and U. S. Air Force support charges, as these costs are highly dependent on mission requirements. The breakdown of costs shown in Table 6 are based on actual or estimated expenses billed to outside agency users such as ESSA, Comsat, and ESRO for reimbursement to NASA and a projection of these costs into the 1972 time frame when the Delta, Model 904 shall be operational. Actual charges for any given mission will, of course, vary to reflect the specific mission requirements.

For launches conducted for outside government agencies and private industry, identifiable launch service charges are segregated and charged directly against the mission. Indirect or cost not identifiable to a peculiar mission are prorated normally over the duration of a launch services contract or a number of Delta launches and allocated accordingly.

F. Future Growth

At this time two major upratings of the Delta are under consideration; however, none are officially approved changes. The first uprating is an adoption of the Saturn 1B stage Rocketdyne H-1 engine to the UBT Thor; an uprating that would benefit Delta and Agena. For a typical geo-synchronous transfer orbit (100 × 19,400 n.mi. at 28° inclination) the increase in the Delta 904 performance, for example, is about + 100 pounds. Two approaches to the use of the H-1 engine

Table 6

Delta Launch Costs
(1972)

	Costs (Thousand Dollars)	
	Initial Launch	Follow-on Launch
HARDWARE		
First Stage Core	1,200	1,200
Thrust Augmentation Solid Motors (9)	630	630
Second Stage and Fairing	1,600	1,600
Third Stage	130	130
Attach Fitting	35	35
LAUNCH SERVICES		
Software/Analysis/Sustaining Support	650	500
Vehicle Checkout		
Production Area	200	200
Launch Site	950	950
RANGE LAUNCH SUPPORT	*	*
TRANSPORTATION	15	15
PROPELLANTS	30	30
NASA ADMINISTRATIVE CHARGES	170	170
TOTAL	$5,610	$5,460

*Range Tracking, data acquisition, technical operations, and U. S. Air Force support charges dependent on mission requirements.

are being considered, (1) modifying the UBT engine section to accommodate the new engine "as is" (2) repackaging the H-1 engine to effect an interface with the UBT identical to the current Rocketdyne MB-3 Block III engine. A decision on use of the H-1 on Thor will be reached by early 1971 to assure booster production continuity into mid-1972.

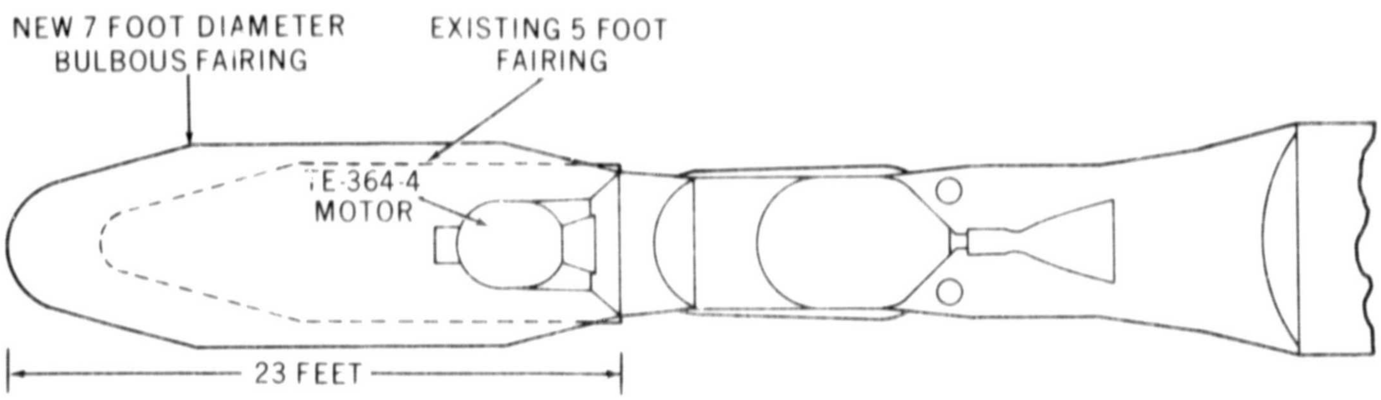


Figure 10. Delta Seven Foot Bulbous Fairing (Under Study.)

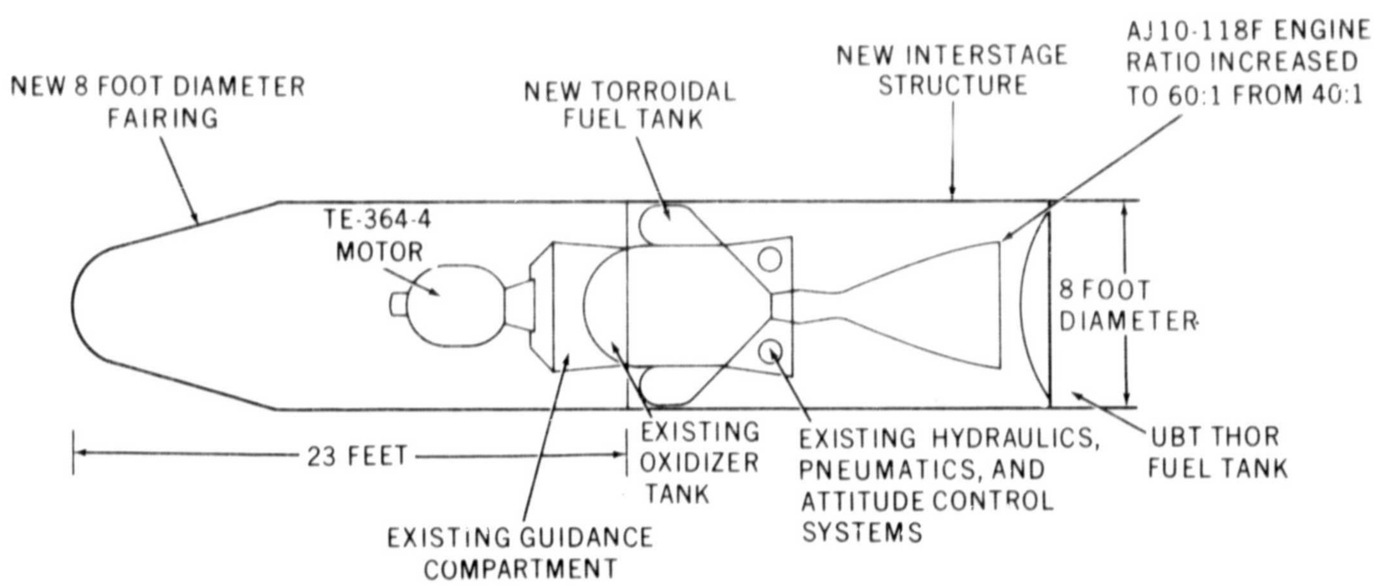


Figure 11. Delta Eight Foot Diameter Fairing and Second Stage (Under Study.)

The second uprating of Delta currently being studied is directed to providing users with a larger spacecraft fairing envelope. Two approaches are being investigated, (1) development of a seven foot diameter bulbous fairing to be used with the current vehicle configuration as shown in Figure 10 and (2) developing on eight foot diameter cylindrical fairing and reconfiguring the Delta second stage propellant tankage to a comparable diameter by replacing the current cylindrical fuel tank by a torroidal tank as shown in Figure 11.

In addition to the larger spacecraft envelop, the second approach has three other advantages, (1) the overall length of the vehicle is reduced about seven feet which enhances the booster steering control margins in high upper atmosphere winds (2) the available volume for attachment of secondary experiments or satellites is dramatically increased both at the forward and aft ends of the stage and (3) the new eight foot diameter interstage permits the second stage engine expansion ratio to be increased to 60:1 from 40:1 which increases the specific impulse about five seconds. For a typical geo-synchronous transfer orbit the change in the Delta Model 904 performance capability, for example, is about -60 pounds for the 7 foot bulbous fairing configuration and about + 100 pounds for the 8 foot cylindrical fairing.

If the adaption of the H-1 engine and/or the development of a larger spacecraft fairing is officially approved at the completion of the current studies it is estimated these upratings would be available for Delta users in late 1972.

II. THOR/AGENA

A. The Evolution of Thor/Agena

The Thor/Agena launch vehicle is the result of over 15 years growth in two of the most versatile and reliable space vehicle stages available—the Thor produced by McDonnell Douglas Astronautics Corp. and the Agena produced by Lockheed Missiles and Space Corp. (LMSC)

Over the past decade the Agena stage boosted by Thor, Atlas and Titan vehicles, has been used by NASA and the Air Force to perform a wide variety of missions in the capacity as both an ascent stage and as a spacecraft with long duration on-orbit capability. To date over 170 Agena stages have been launched.

At the initiation of the U. S. space effort, two significant steps were taken to assure ultimate long-term capability in the intermediate class of boosters. One was the beginning of development of the Thor IRBM in 1955 and the other was initiation of the Agena in 1956. Conceptually, Agena was to be the second stage

for the Atlas booster; however, upon development of both the Thor and Agena along almost parallel milestones, it was inevitable that they should be merged to satisfy the accelerating demand for high-performance space boosters. The earliest version was the Thor/Agena A. This vehicle was a modified Thor IRBM and a short tank Agena; in February 1959 it placed 1300 pounds into a 100 by 700 n. m. orbit—the first polar launch from the Western Test Range. The spacecraft was designated "Discoverer I" and became the first of thirty eight flights carrying this designation.

In 1960 Agena B was launched with a modified Thor to become the second generation vehicle within this class. Agena B had more advanced electrical and guidance equipment as well as larger propellant tanks. In the meantime Agena A and Agena B was teamed with Atlas and NASA used an Atlas/Agena B design for its Ranger program.

In 1961 the U. S. Air Force began development of a newer version of the Agena; the Agena D. This vehicle had all the versatility of a building block system with configuration control; it became the Standard Agena. Agena D was the result of new design techniques, employing optimum structures and solid-state avionics. New magnesium and beryllium materials were used and open-tube tubular structures provided lower weights than the Agena B; further significant payload increases were derived from the dual-burn engine. The pressurization system for the propellant tanks was changed to an orifice feed and an advanced propellant scavenging and containment system was incorporated. In 1962 Thor boosted Agena D into polar orbit thus becoming the third generation of this combination.

NASA continued use of the Atlas/Agena vehicle until 1964 when the Echo 2 inflatable satellite was placed into a 600-800 n.mi. near polar orbit. The following year the Thor/Agena D launched the first of the Orbiting Geophysical Observatory missions. The OGO program was completed in 1969 with the launch of OGO-6; half of the flights were on the Atlas/Agena family and half were on the Thor/Agena family. In addition to the OGO missions, PAGEOS, Explorer 31, Nimbus and SERT used the Thor/Agena combination. Other NASA programs exploited the higher capability of the Atlas/Agena family; these programs included the five Lunar Orbiters, three ATS's, five Mariners, nine Ranges, six Gemini Targets vehicles and an Orbiting Astronomical Observatory (OAO).

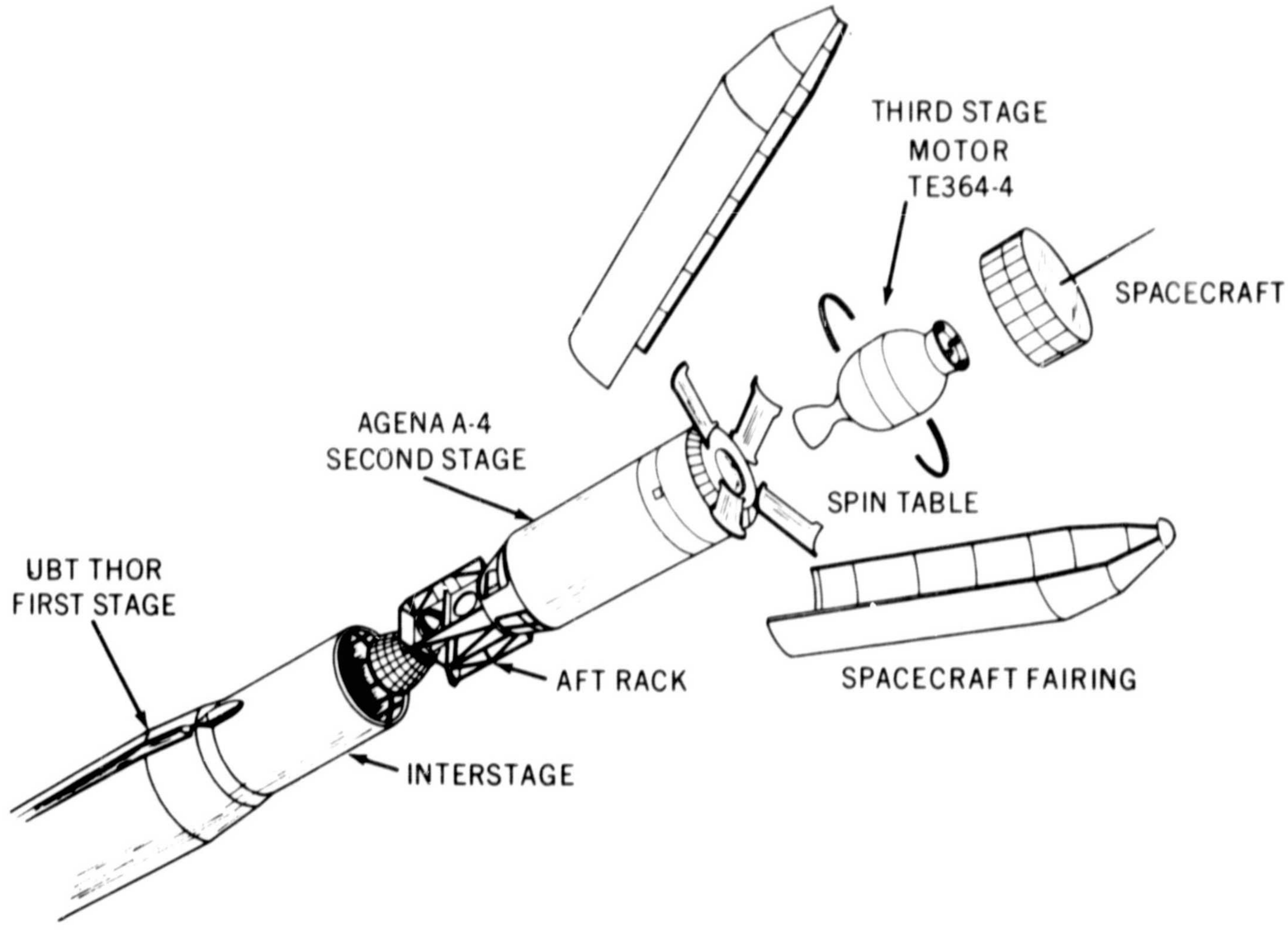
Today a new Thor/Agena, Model 9A4, three stage ascent launch vehicle with higher performance than the Delta can be assembled by using in place of the Delta second stage the uprated Agena second stage, Model A-4, now under development by the U. S. Air Force for use on the Titan booster. Currently, however, neither NASA nor the Air Force have scheduled any missions for the Thor/Agena, Model 9A4 configuration. Hence, the first user must sustain a non-recurring

cost for vehicle integration and launch facilities modifications necessary to bring the Thor/Agena into NASA's inventory of vehicles. In order to provide mission planners with data on this new combination the Thor/Agena, Model 9A4, is described together with its performance, flight environment, and launch costs.

B. Vehicle Description

The three stage Thor/Agena, Model 9A4 shown in Figure 12 stands 110 feet and weighs approximately 266,000 pounds at lift-off. The launch vehicle is designed for ascent through 95% ETR and WTR upper atmosphere annual wind profiles, lift-off in 43 knot ground winds, and hold on the launch with all propellants loaded for several hours in readiness for short launch window requirements.

The first stage UBT Long Tank Thor is essentially identical to that used to boost the Delta vehicle. As on Delta the first stage guidance and control originates from the Agena A-4 second stage inertial strapdown guidance system developed by Minneapolis-Honeywell. A 8192 word memory digital on-board guidance



computer translates linear and angular acceleration data obtained from body-mounted accelerometers and gyros into velocity and position data in the navigation coordinate system. The explicit navigation and steering equations, supplemented by error compensation and logic equations, control the guidance discrete flight functions and pre-programmed command attitude rates from launch through spacecraft separation.

The interstage section between first and second stage is provided with guide rails, a pair of retro rockets and a mild detonating fuse separation joint between the top of the interstage and the second stage. Sixteen seconds after first stage burnout the joint is detonated and the retro rockets are ignited. The booster is backed away and the Agena is guided out of the interstage on the rail assembly. Nineteen seconds later the Agena engine is started.

The Agena Model A-4 second stage, an advanced version of the Agena D, is 20 feet long, 5 feet in diameter and weighs 17,500 pounds at ignition. The Bell Aerosystems pump fed engine develops 16,000 pounds thrust and operates for 240 seconds on UDMH and IRFNA propellants. Although two-burn capability is standard, single- and three-burn options are available, using special start-can adapters.

The stage consists of four major structural sections; the forward rack, propellant tank section, the aft rack and booster adapter. These section provide for the mounting and installation of the equipment that makes up the propulsion, electrical, guidance, and telemetry and tracking subsystems. The Agena structure is designed to allow installation of secondary payload experiments in both the forward and aft equipment racks. Up to 1,000 pounds of experiments and mission support equipment can be carried on the aft rack. A variety of flight proven secondary experiment mounting assemblies for retained and ejectable payloads have been flown in this location as well as on the forward interface ring.

The Agena forward section contains the inertial strapdown guidance system, flight controls electronics, pyro-operated helium valve for propellant pressurization, electrical power, telemetry and tracking and provides additional mounting space for other equipment which may be desired on the flight. The tank section serves as the aerodynamic surface and supporting structure between the forward and aft sections. It contains the fuel and oxidizer required to operate the main propulsion system in a dual compartment tank which includes containment and scavenging features to maintain proper propellant orientation for multiple engine firing in the zero "g" space environment.

During the Agena second stage powered flight, pitch and yaw steering is provided by gimballing the engine and roll is controlled by cold nitrogen gas jets. During

coast phases, cold nitrogen gas jets control the vehicle in all axes. The control system electrical power and nitrogen gas system is capable of maintaining second stage attitude for twenty hours; however, modification permits extending this period. The vehicle is capable of being reoriented or may be placed in a controlled tumbling mode to accommodate spacecraft thermal requirements on long coast missions. This feature, as well as other maneuvering modes, is implemented in the computer programming without hardware changes. In addition, the Agena A-4 flight control system may be configured to provide long-term, low-power, low-gas-usage orbit attitude control. This is accomplished by adding a horizon sensor or star tracker reference and summing these data with gyro data in the flight control electronics. Power may then be removed from the rest of the guidance system.

The Thor/Agena has been used frequently by NASA and the Air Force for placing secondary experiments and satellites into orbit. Examples are the Distance Measurement Experiment (Explorer 31) on the ISIS mission and the Sequential Correlation of Range (SECOR) experiment on the Nimbus B-2 mission. The Agena has been used for over 40 flights of multiple payloads; in addition, a Small Research Satellite (SRS) has been developed specifically to use the piggy-back potential. The first SRS was launched in 1963 as a secondary payload from an orbiting Agena space vehicle; since that time 20 such subsatellites have performed similar scientific missions successfully. Such multiple payloads can be placed into operation simultaneously with a single Thor/Agena without compromising or interfering with the identity or objectives of the prime mission. The inherent reliability of the Agena stage, developed and extensively used as a long-life spacecraft, make it an attractive candidate as a dependable on-orbit platform.

The third stage spin table, TE-364-4 solid motor and spacecraft attach fitting assemblies are identical to those used on Delta and have been previously described. For the two stage configuration the Agena provides a common interface for spacecraft users at the forward ring.

Several spacecraft shrouds are developed and flight proven for use on the Agena. The Delta shroud previously described and shown in Figure 6 may be used on Agena. In addition, there are the Standard Agena Clamshell Shroud (SACS) and the Light Weight Clamshell Shroud (LCS). Both have transitions that either accepts a spacecraft adapter or a third stage spin table assembly. A wide variety of flight proven spacecraft adapters are available to accommodate separating spacecraft. The SACS which has been used with OGO, ATS, and Nimbus spacecraft is RF transparent and 18-1/2 feet long. It is fabricated with a fiberglass shell, internal metal frames, and longerons. The LCS is an all-metal, flight proven shroud available in lengths from 10 to 30 feet in increments of 5 feet, using standard lengths. The SACS half shells are strapped together with stainless steel

tension bands that are cut to allow shroud separation. The LCS employs a pyrotechnic seam fracture separation system that divides the shroud into half shells as it cuts the shroud longitudinally and circumferentially at the separation plane. The pyrotechnic residue is contained to prevent smoke and debris from contaminating the spacecraft. The configuration permits off-pad spacecraft encapsulation. The spacecraft envelope and shroud dimensions are shown in Figure 13 for the SACS and Figure 14 for the LCS.

C. Flight Sequence and Performance

The Thor/Agena Model SA4 flight profile for a three stage synchronous transfer mission is similar to that shown in Figure 7 for the Delta vehicle. The flight profile for a two stage sun-synchronous 600 n. m. circular orbit representative of the Nimbus mission is shown in Figure 15. The vehicle is launched from WTR on an azimuth of 194 degrees. The Agena second stage is injected into a 100 by 600 n.mi. Hohmann transfer and coasts 180 degrees around the earth to apogee where the Agena restarts and circularizes the orbit. After spacecraft separation the spent Agena stage retros out of the spacecraft orbit to preclude a collision should it be accelerated by venting pressurants or propellants.

The Thor/Agena three sigma injection dispersions for the 600 n.m. sun-synchronous mission is shown in Table 7. The high accuracy is obtained by the strapdown inertial guidance system the Agena A-4 model employs in contrast to all earlier models of Agena that utilized an autopilot augmented by radio guidance.

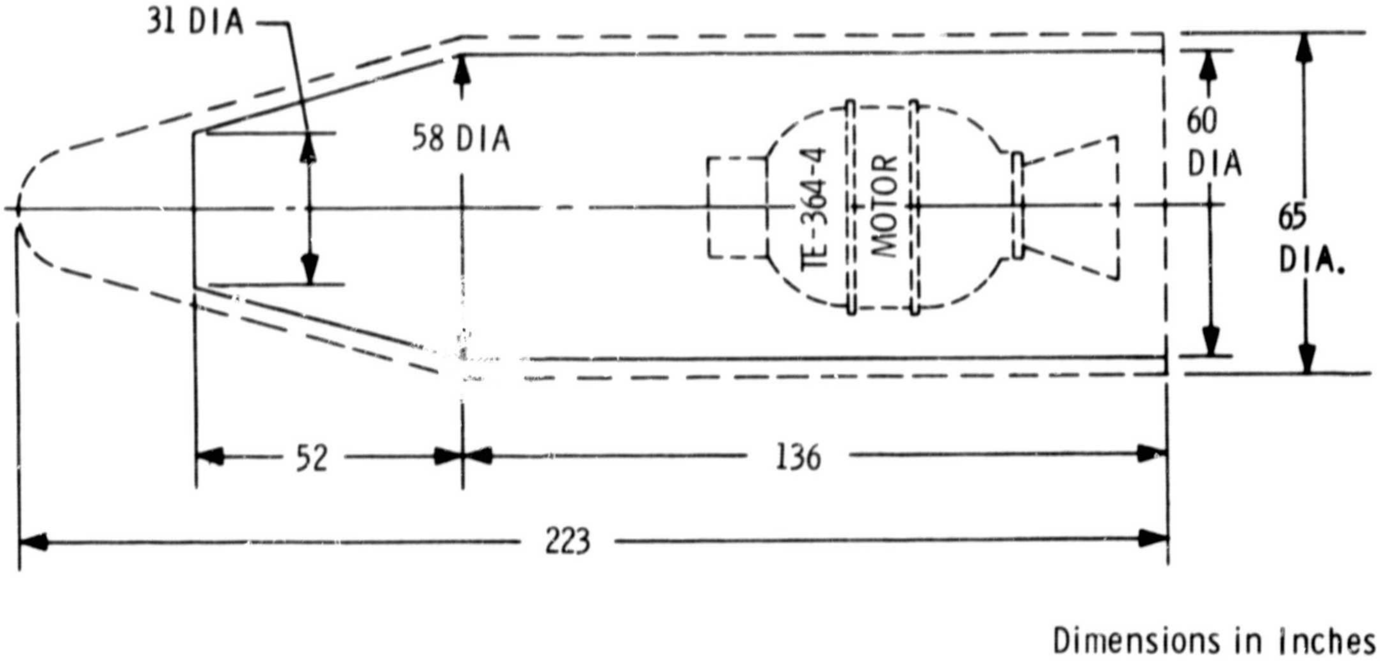


Figure 13. Standard Agena Clamshell Shroud (SACS.)

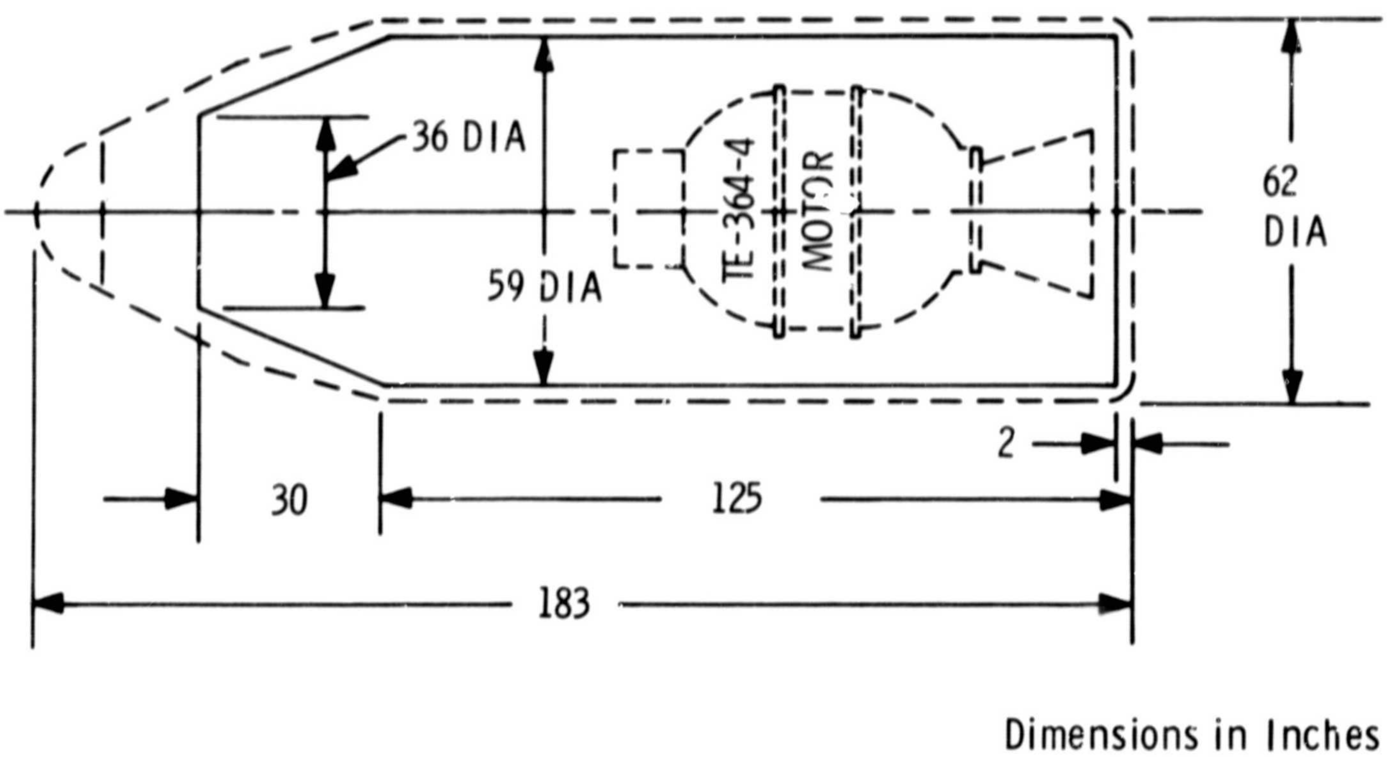


Figure 14. Lightweight Clamshell Shroud (LCS.)

NO.	TIME (SEC)	EVENT
0	0	LIFTOFF
	85	JETTISON SOLID MOTORS
	220	FIRST STAGE BURNOUT
3	256	SECOND STAGE IGNITION
	266	FAIRING JETTISON
5	487	SECOND STAGE CUTOFF
6	494	BEGIN EARTH-RATE PITCH
7	3261	SECOND STAGE RESTART
8	3267	SECOND STAGE CUTOFF
	3517	SPACECRAFT SEPARATION

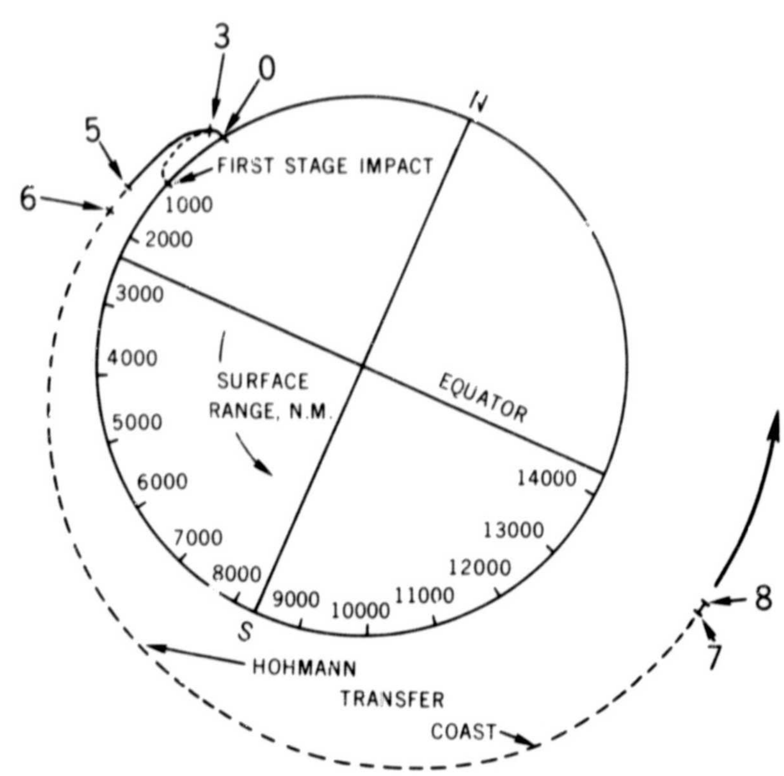


Figure 15. Thor/Agena Flight Sequence of Events for a Polar Circular Mission.

Table 7

Sun-Synchronous Orbit Dispersions

Parameter	Nominal	99% Probable Dispersions	
		MIN.	MAX.
Apogee Altitude, N.M.	600	-5	16
Perigee Altitude, N.M.	600	-17	4
Orbital Period, minutes	107.4	-0.4	0.4
Orbit Eccentricity	0.000	000	0.002
Orbital Inclination, degrees	100	-0.07	0.07

The payload weight versus characteristic inertial velocity for two and three stage Thor/Agena launches from ETR and WTR is shown in Figures 16 and 17. As before the first number in the model designation (3, 6 or 9) indicates the number of thrust augmentation solid motors on the UBT Thor, the digit (A) the Agena A-4 second stage with inertial guidance, and the third digit (4) notes the -4 version of the TE-364 third stage motor.

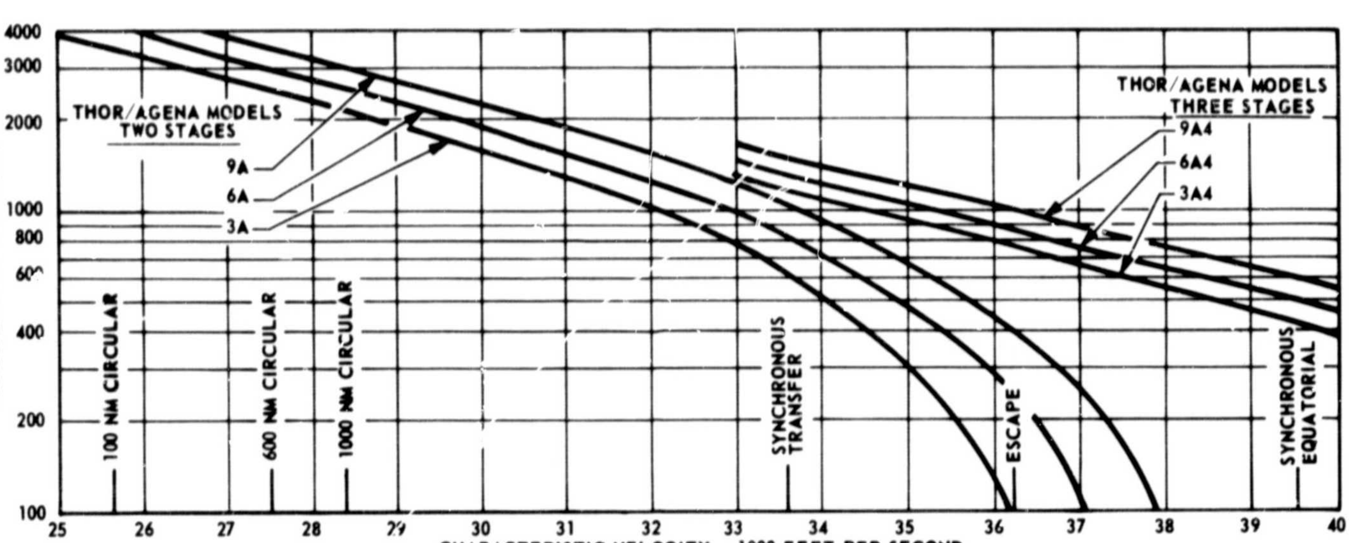


Figure 16. Thor/Agena Payload Capability-Eastern Test Range.

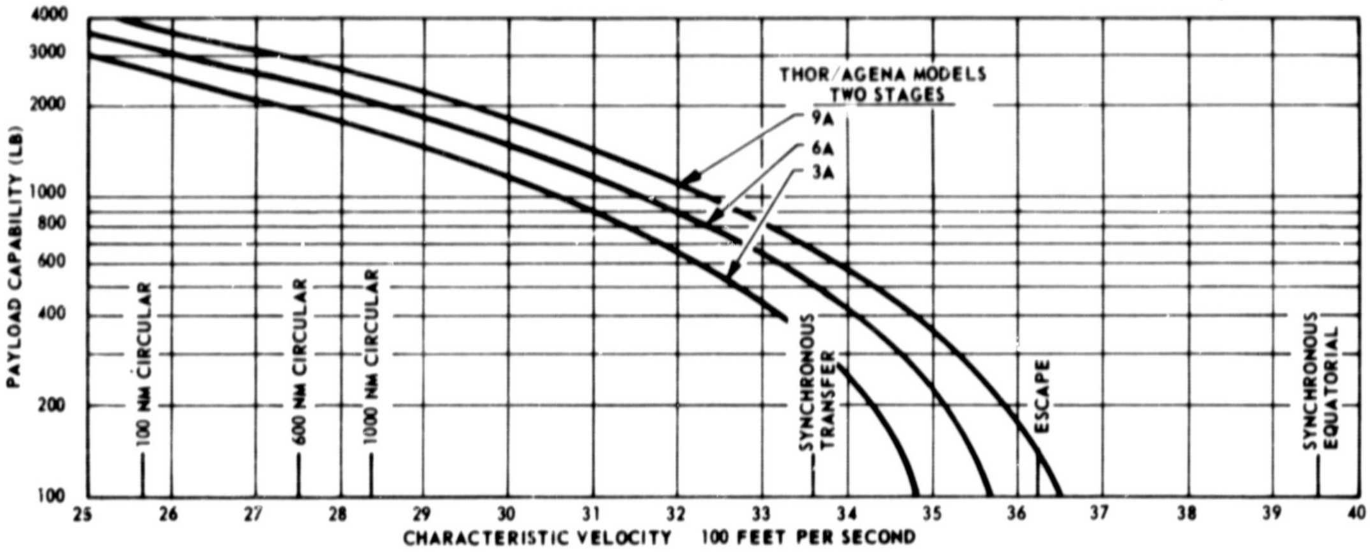


Figure 17. Thor/Agena Payload Capability-Western Test Range.

D. Flight Environment

The environment imposed on the spacecraft by the Thor/Agena is estimated from previous flight measurements. A summary of the expected critical flight environment is provided in Table 8 for use in preliminary studies by spacecraft mission planners. The Thor/Agena environmental forcing functions are the same as those described for the Delta critical flight environment.

E. Costs

The projected cost of Thor/Agena, Model 9A4, reimbursable launches in 1972 is about $7.0 million dollars. However, this cost is highly dependent upon the Agena stage production rate sustained at that time by the Air Force, currently the sole user, and the availability of Air Force check out and test equipment. The cost of the last NASA Thor/Agena launch (Nimbus D) in April 1970 was about $6.5 million. The approximate breakdown of these costs in Table 9 are estimates projected into the 1972 time frame and assumes the Air Force usage is six Agena stages per year and a three vehicle program for NASA users can be added on to this production base. In any event actual charges for any given Thor-Agena vehicle program will, of course, vary to reflect the specific program constraints and requirements.

In addition to the reimbursable launch costs, the first user of the Thor/Agena 9A4 would be charged an estimated one time cost of about $3 million to modify the existing Air Force Agena design and software to accommodate the UBT Thor

Table 8

Thor/Agena Critical Flight Environment

Excitation	Flight Event	Duration	Two Stage		Three Stage	
		Seconds	Frequency, Hz	Level	Frequency, Hz	Level
Sinusoidal Vibration Thrust Axis	Lift Off T + 210 sec.	2 to 5 5 to 7	10-17 17-23 23-100	1.5g (O-P) 4.5 1.5	10-17 17-23 23-100	1.5g (O-P) 5.0 1.5
Lateral Axis	Lift Off	2 to 5	5-14 14-100	1.3* 1.0	5-14 14-100	1.5* 1.0
Random Vibration Three Axis	Transonic & Max. Q	10 to 15	20-300 300-1000 1000-2000	+ 4db/octave 0.06^2/Hz -6db/octave	20-300 300-2000	+ 3db/octave $0.02g^2$/Hz
Shock	Fairing Separation	0.001	1200g at 0.8 milliseconds Terminal Peak Saw Tooth		1400g at 0.3 milliseconds Terminal Peak Saw Tooth	
Steady State Acceleration	First Stage Burnout			6.7	6.0	
	Third Stage Burnout				23.5g for 500 lbs. spacecraft 10g for 1500 lbs. spacecraft	
Acoustic	Lift Off and Transonic	10 to 15	142db 37 to 9600 Hz (peak level 800 to 1000 Hz)			

*For spacecraft weight of 1000 pounds. For lighter spacecraft, levels are higher.

Table 9

Thor/Agena Launch Costs
(1972)

	Costs (Thousand Dollars)	
	Initial Launch	Follow-On Launch
HARDWARE		
First Stage Core	1,200	1,200
Thrust Augmentation Solid Motors (9)	630	630
Second Stage and Fairing	2,600	2,600
Third Stage	130	130
Attach Fitting	35	35
LAUNCH SERVICES		
Software/Analysis/Sustaining Support	750	600
Vehicle Checkout		
Production Area	420	420
Launch Site	1,200	1,200
RANGE LAUNCH SUPPORT	*	*
TRANSPORTATION	15	15
PROPELLANTS	30	30
NASA ADMINISTRATIVE CHARGES	200	200
TOTAL	$7,210	$7,060

*Range Tracking, Data Acquisition, Technical Operations and U. S. Air Force Support Charges dependent upon mission requirements.

booster and a third stage and to integrate the vehicle and modify the support equipment and facilities at ETR. This charge is about $1 million less for a launch from WTR.

F. Future Growth

The Agena performance is being uprated by addition of 44% N_2O_4 to the IRFNA oxidant. This denser, higher performance propellant that increase the specific

impulse of the engine by approximately seven seconds is to be flight demonstrated this year. For a typical geo-synchronous transfer orbit the increase in Thor/Agena 9A4 performance, for example, is about 150 pounds.

III. ORGANIZATION AND INTERFACES

Delta and Agena users interface organizationally with three elements within NASA. This is best illustrated by the relationship that existed between the European Space Research Organization's, Project HEOS, and the NASA Delta-Agena Project on the Delta 61 launch as is shown in Figure 18.

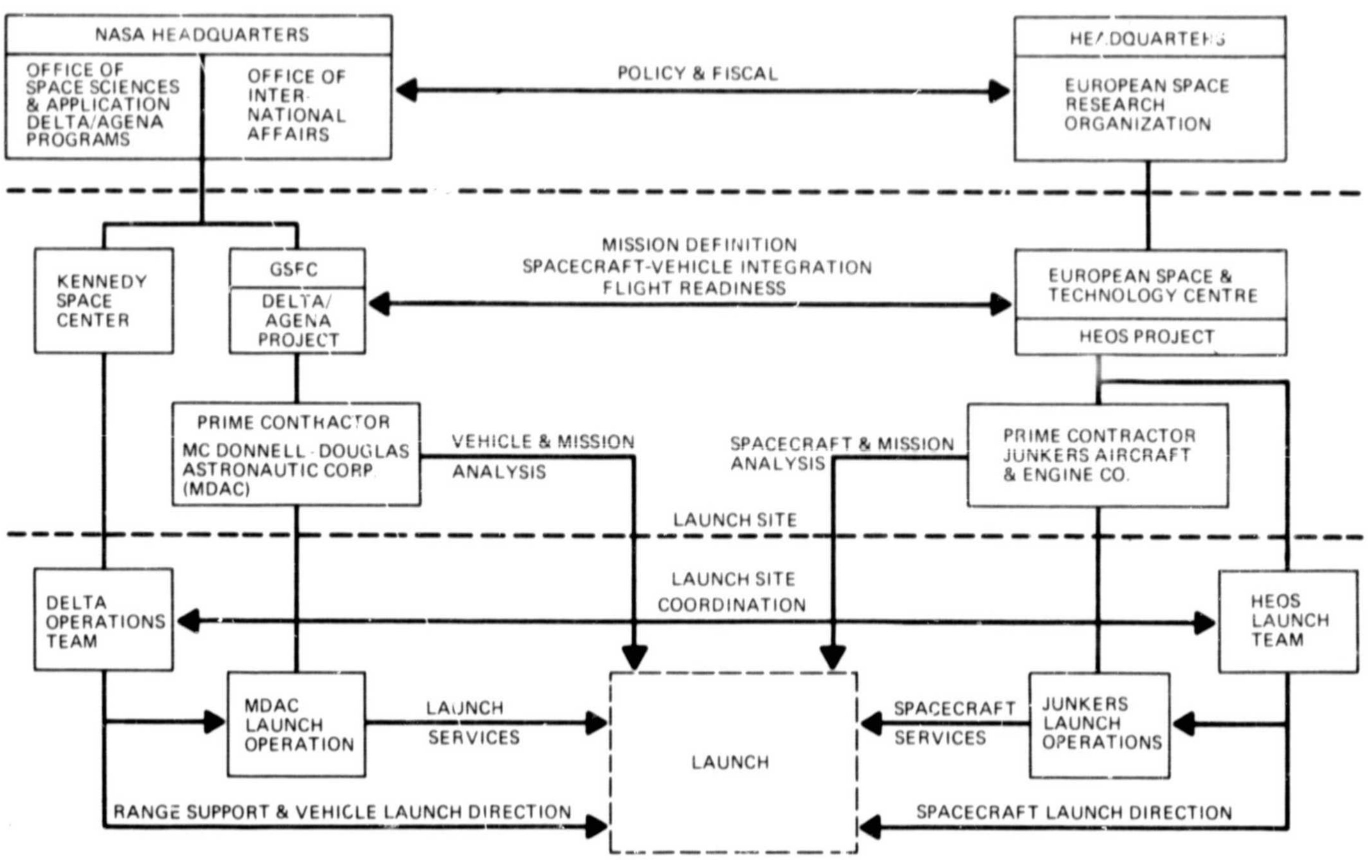


Figure 18. Organization and Interfaces

Agreement between NASA and a foreign space organization for a launch and associated services is established at NASA Headquarters level. This is normally done with a Memorandum of Understanding outlining the principles under which such arrangements are to be made, followed by a specific contract for each mission. The agree-to policies and fiscal arrangements are then passed through the NASA Office of Space Sciences and Applications (OSSA) Delta-Agena Programs Office to the Goddard Space Flight Center (GSFC) Delta-Agena Project Office for

implementation. The Delta-Agena Project Office is vested with the authority and responsibility for carrying out all aspects of a Delta or Agena vehicle mission. The Delta-Agena Project works directly with the Spacecraft Project to develop and define the spacecraft/vehicle mission requirements, integrate the spacecraft to the vehicle, establish schedules, and determine the final flight readiness of the vehicle. The Delta-Agena Project contracts with and directs an industrial contractor for vehicle hardware, mission analysis, and launch support services; McDonnell-Douglas Astronautics Corporation (MDAC) for the UBT Thor and Delta and Lockheed Missles and Space Coporation (LMSC) for the Agena. Direction of the launch support services furnished by MDAC or LMSC at the launch site is delegated to the NASA Kennedy Space Center (KSC). The KSC Delta Operations Team works directly with the Spacecraft Project at the launch site to insure required Range and contractor services are provided and to coordinate the launch site vehicle and spacecraft activities.

This simple organizational structure with short and direct authority and communications lines is a significant factor in the flexibility and responsiveness Delta and Agena can provide its users.

IV. SPACECRAFT INTEGRATION AND LAUNCH OPERATION

Delta or Agena vehicle interface constraints together with performance and accuracy estimates are provided to potential vehicle users as soon as the concept of the mission is outlined to the NASA, Goddard Space Flight Center Delta-Agena Project Office. The Delta-Agena Project welcomes and encourages early definition of prospective missions by potential users. In some instances, mission definition and integration planning has preceded actual mission commitment by two and three years. Experience has demonstrated that this advance and continuous coordination between the user and the Delta-Agena Project during the period of developing mission requirements, enhances the visibility of both parties and reveals problem areas before final definition of the spacecraft/vehicle interface and trajectory parameters. In general, spacecraft/vehicle planning for new missions follow the pattern and time frame outlined in Figure 19 and starts about one year (T-52 weeks) before launch when the Spacecraft Project provides the Preliminary Mission Definition and Requirements to the Delta-Agena Project Office. This definition encompasses the preliminary spacecraft configuration, mass properties, trajectory, and orbital requirements necessary for preliminary vehicle performance evaluation and analysis. A preliminary trajectory with attendant injection error studies and thermal studies is completed within ten weeks. With this visibility, the Delta-Agena Project and the Spacecraft Project jointly develop a Final Mission Requirement specification (T-40 to T-26 weeks) that includes such constraints as spacecraft orbital lifetime, apogee and perigee altitude

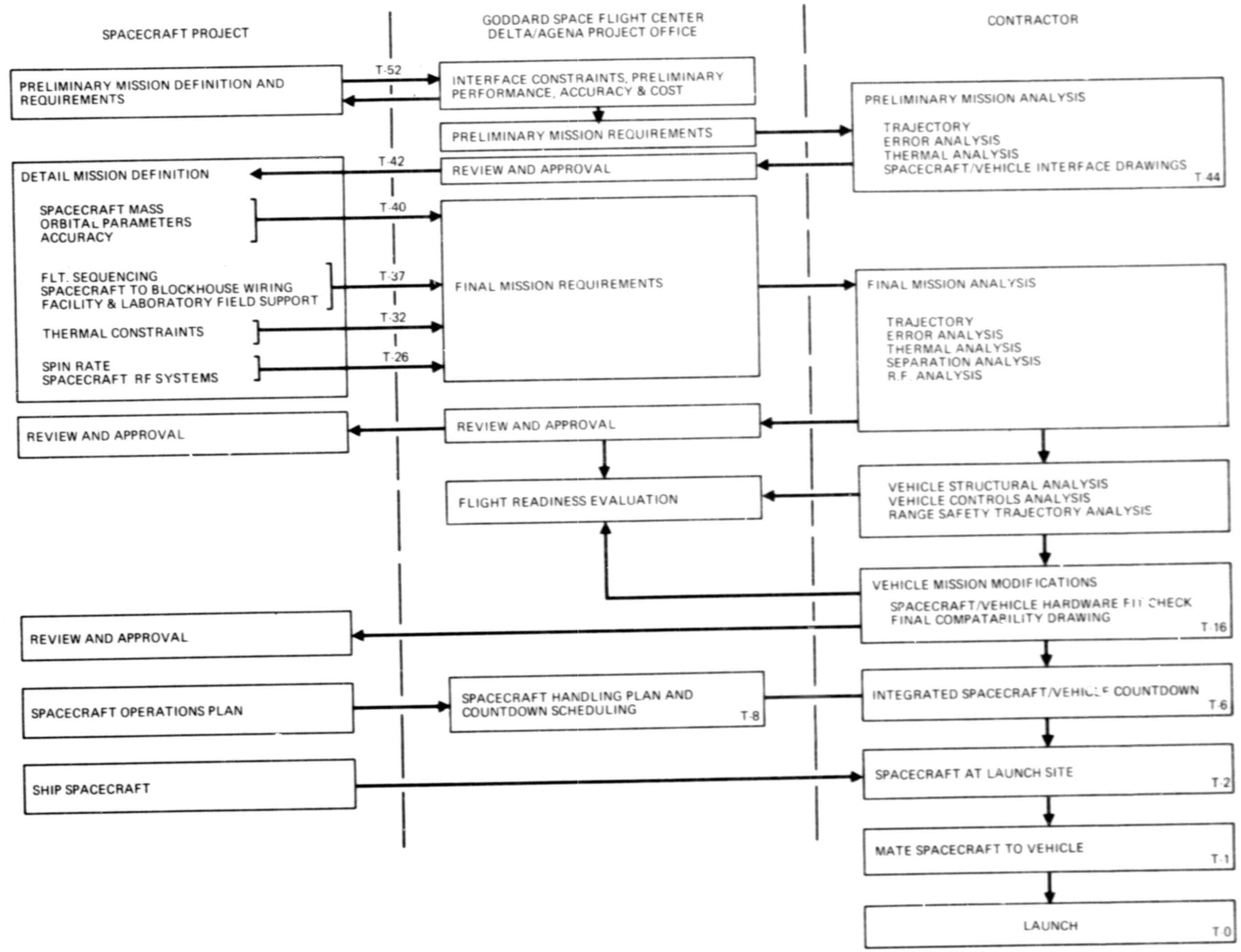


Figure 19. Delta/Agena Mission Analysis and Integration

and geocentric location, permissible injection errors, injection attitude orientation, launch window criteria, tracking and data retrieval requirements, spacecraft mass properties, and all other data necessary for the preparation of the Final Mission Analysis.

The Spacecraft Project reviews the final mission trajectory about T-35 weeks and the final injection and orbital error analysis about T-23 weeks. The trajectory includes all technical data defining the flight mode, sequence of flight events, vehicle weights and propulsion system characteristics, tabulations of trajectory parameters, weight history, radar look angles, and instantaneous impact loci. Final definition of the maximum and minimum allowable spin rate, spacecraft RF systems, and permissible inflight thermal inputs are provided to the Delta-Agena Project by T-26 weeks. A full scale compatibility drawing based on the Spacecraft Project's final configuration drawings is prepared normally at T-16 weeks. This drawing is primarily to show all clearances between the spacecraft and fairing, attach fitting, and third stage motor and locate the orientation of such features as umbilical connectors, access ports through the fairing, and any special interface wiring between the attach fitting and spacecraft. A Spacecraft Handling Plan is jointly developed and finalized about T-8 weeks and describes all hazardous systems, spacecraft test procedures, and details pre-launch work schedules. Typically the spacecraft arrives to the launch-site two weeks before launch (T-2) and is built up on the third stage motor assembly the following week and mated to the vehicle on the pad one week before launch for RFI testing with the vehicle and Range RF systems. Final weights are inputed to trim the final trajectory parameters in the inertial guidance computer the week of launch.

For three stage missions the spacecraft must be statically and dynamically balanced prior to receipt at the launch site. The allowable spacecraft center-of-gravity offset and principle axis misalignment is 0.015 inches and 0.002 radians, respectively. For missions where injection attitude is extremely critical for mission success, a third stage assembly composite spin balance is conducted at the launch site.

The Delta-Agena Project conducts launches from both ETR and WTR. Prograde missions with orbital inclinations of 30 degrees or less are normally launched from ETR and near-polar or retro-grade missions from WTR, though near polar missions have been launched from ETR on Delta.

Facilities for the Spacecraft Project use at the launch site include spacecraft assembly and checkout laboratories, telemetry, fabrication and cryogenic laboratories, clean rooms, shops, storage, and offices.

The first and second stage mission modifications to the vehicle are made in the contractor production area. The first and second stages are delivered directly

to the launch pad, erected, and again undergo systems testing. The thrust augmentation solid motors and third stage solid motors are stored and prepared at the launch site. The thrust augmentation solid motors are mated to the first stage on the launch pad about two weeks before launch. The third stage motor is built up on the spin table and the spacecraft mated with the assembly at about the same time. The spacecraft/third stage assembly is transported in an environmentally controlled canister to an environmental room on top of the mobile service tower around the vehicle and there the assembly is mated to the vehicle. While the spacecraft is mated to the vehicle, spacecraft and vehicle checkout and testing is interspersed and whenever possible to accommodate the spacecraft requirements.

On pad checkout of the vehicle culminates in a pre-countdown simulated flight without propellant on-board, wherein all systems of the vehicle are exercised as they are during the mission. The simulated flight test takes place one week before launch and is followed by final preparation of the vehicle for launch and then a three day countdown to lift-off. If necessary, complete access to the spacecraft can be provided up to four hours prior to liftoff, though normally the fairing is installed about 12 to 16 hours prior to launch. Provisions to continuously power and monitor the spacecraft from the blockhouse is provided through the vehicle wiring. While the spacecraft is on the vehicle, thermally and hermetically conditioned, filtered air is provided to the spacecraft right up to lift-off.